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Continued Evaluation and Spectrum Development of a Health and Usage Monitoring System

May 2004

Final Report

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16. Abstract

This report describes the results of a research project to evaluate structural usage monitoring and damage tolerance methodology using data collected on a Bell Model 412 helicopter that was equipped with a health and usage monitoring system (HUMS). The helicopter was operated by Petroleum Helicopters Inc., from its base in Morgan City, Louisiana. The operational mission for this helicopter was referred to as the utility mission in Morgan City (UMMC). The usage spectrum data from the UMMC were compared to certification data and to data from previous HUMS studies. The previous HUMS studies included the Atlanta short haul mission (ASHM), which was considered severe usage because it involved many short maneuvering flights to provide pickup and delivery services at the Olympics, and the Gulf Coast mission (GCM), which was considered mild usage because it primarily involved long cruise flights. The usage spectrum for the UMMC was, in general, more severe than the GCM but less severe than the ASHM.

The results of the different missions showed that usage monitoring could provide benefits in extending retirement times or inspection intervals compared to those set during certification, especially if high- and low-altitude effects are considered. In addition to usage monitoring evaluations, simplified mini-HUMS approaches were reviewed that could potentially provide low-cost HUMS with high paybacks. Also, guidelines were developed for HUMS certification and qualification and for the integration of HUMS into the operator's maintenance program. These guidelines are also discussed in the report. Finally, the usage data that were collected were used to perform fatigue-life calculations and theoretical redesigns of four selected rotor system components known as principal structural elements to meet damage tolerance requirements. This theoretical damage tolerance investigation evaluated the weight impact and practicality of damage tolerance fatigue methodology versus a safe-life methodology.

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LIST OF ACRONYMS AND SYMBOLS

ASHM	Atlanta short-haul mission
BHTI	Bell Helicopter Textron Inc.
COTS	Commercial Off-The-Shelf
FAA	Federal Aviation Administration
FCR	Flight condition recognition
GAG	Ground-air-ground
GCM	Gulf coast mission
GPS	Global positioning system
GW	Gross weight
H_d	Density altitude
H_p	Pressure altitude
HUMS	Health and usage monitoring system
IGE	In-ground effect
MMS	Maintenance management system
PC	Personal computer
PHI	Petroleum Helicopters Inc.
PSE	Primary structural element
rpm	Revolutions per minute
UMMC	Utility mission in Morgan City
URD%	Unrecognized damage percentage
V_h	Maximum horizontal velocity
V_{ne}	Velocity never to exceed

EXECUTIVE SUMMARY

This report describes the results of a research project to evaluate structural usage monitoring and damage tolerance methodology using data collected on a Bell Model 412 helicopter equipped with a health and usage monitoring system (HUMS). The helicopter was operated by Petroleum Helicopters Inc., from its base in Morgan City, Louisiana. The operational mission for this helicopter was referred to as the utility mission in Morgan City (UMMC). The usage spectrum data from the UMMC were compared to certification data and to data from two previous HUMS studies. The first of these previous HUMS studies was the Atlanta short haul mission (ASHM), which was considered severe usage because it involved many short maneuvering flights to provide pickup and delivery services at the Olympics. The second study was the Gulf Coast mission (GCM), which primarily involved long cruise flights and was consequently considered mild usage. The usage spectrum for the UMMC was, in general, more severe than the GCM but less severe than the ASHM.

The results of the different missions showed that usage monitoring can provide benefits in extending retirement times (or inspection intervals) compared to those set during certification, especially if high- and low-altitude effects are considered. In addition to usage monitoring evaluations, simplified mini-HUMS approaches were reviewed that could potentially provide low-cost HUMS with high paybacks. Also, guidelines were developed for HUMS certification and qualification and for the integration of HUMS into the operator's maintenance program, and these guidelines are discussed in the report. Finally, the usage data that were collected were used to perform fatigue-life calculations and theoretical redesigns of four selected rotor system principal structural elements in order to meet damage tolerance requirements. This theoretical damage tolerance investigation evaluated the weight impact and practicality of damage tolerance fatigue methodology versus a safe-life methodology.

1. INTRODUCTION.

This report describes the results of a research program funded by the Federal Aviation Administration (FAA) William J. Hughes Technical Center, Atlantic City International Airport, NJ, to evaluate structural usage monitoring and the damage tolerance methodology using data collected concurrently with a helicopter flight program. This effort was conducted by Bell Helicopter Textron Inc. (BHTI). Usage spectra data (percent time at condition) were collected on a Bell Model 412 helicopter equipped with a health and usage monitoring system (HUMS) and a data recorder. The helicopter was operated by Petroleum Helicopters Inc. (PHI) during continued operations in Louisiana and surrounding areas in the summer and fall of 1999.

Usage data were collected for a utility mission that primarily involved cruise flights along with some shorter flights for pickup and delivery services. This mission is referred to as the utility mission in Morgan City (UMMC) since the helicopter was operated from PHI's base in Morgan City, Louisiana. Usage data recorded during the period were furnished by PHI to BHTI for analysis. See the data analysis flow shown in figure 1-1.

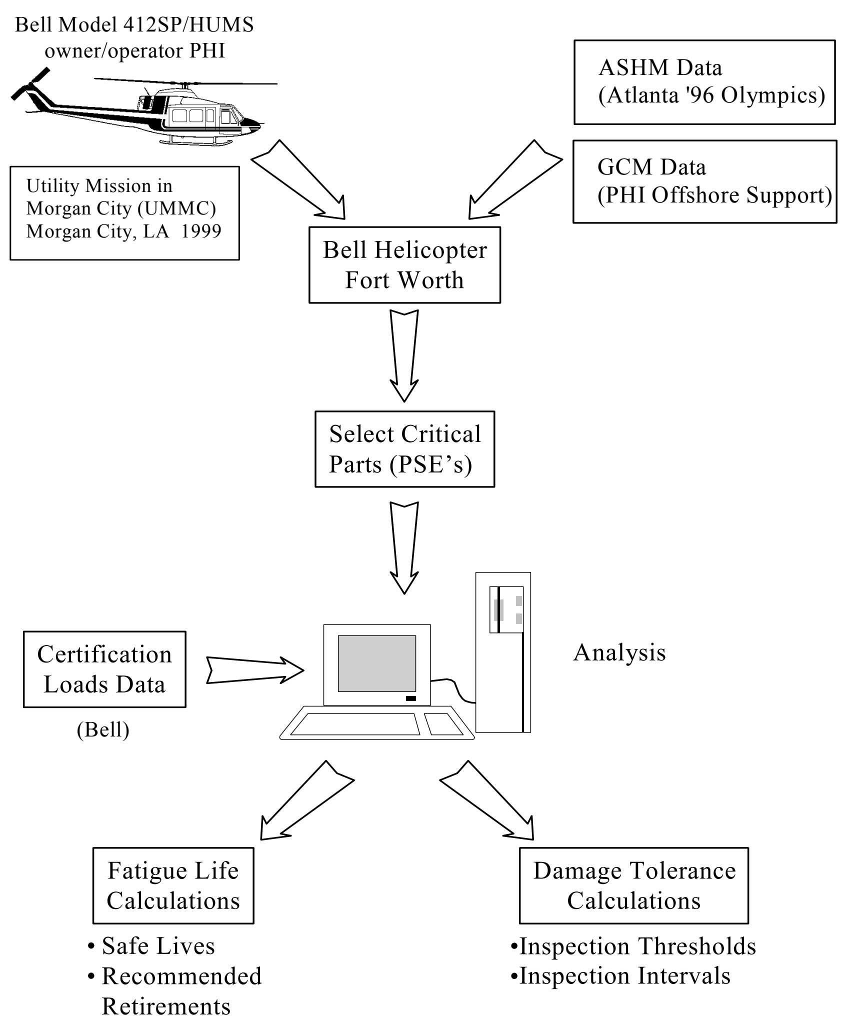

FIGURE 1-1. DATA COLLECTION AND ANALYSIS FLOW

Usage spectrum data from the UMMC were compared to usage spectrum data from two other missions using data collected under two previous FAA-funded HUMS studies. The first of these was the Atlanta short haul mission (ASHM) [1], which involved many short flights to provide pickup and delivery services during the 1996 Olympic Games; the second was the Gulf Coast mission (GCM) [2], which involved long level flights at cruise airspeed for offshore oil support.

An analysis was conducted that involved the calculation of fatigue crack growth in selected critical dynamic components, referred to as principal structural elements (PSEs), based on the usage spectra for the UMMC, ASHM, and GCM, along with certification missions. Based on these calculated results, the PSEs were theoretically redesigned to meet damage tolerance requirements with acceptable inspection intervals. The weight impact and practicality of damage tolerance fatigue methodology versus a conventional safe-life fatigue method were then evaluated.

It should be noted that the lives and inspection intervals developed for purposes of this study are hypothetical and should not be used to draw conclusions concerning design, operational, or maintenance requirements of the Model 412 helicopter. They are meant purely for heuristic purposes only.

The results that were obtained in this project are described in the following sections of this report:

- Section 2 compares the UMMC mission spectrum with other mission spectra.

- Section 3 describes the four PSEs that were selected for analysis.

- Section 4 discusses the results of the fatigue-life analysis of the selected PSEs with comparisons drawn between the UMMC, the ASHM, the GCM, and the certification mission. Also included in section 4 is a comparison of the control boost tube loads from the certification load-level survey against the boost tube loads measured during the ASHM mission. The certification boost tube loads are compared to the ASHM-measured loads using the ASHM and GCM spectra as measured by the HUMS unit.

- Section 5 discusses reduced complexity alternatives or mini-HUMS methodologies that might be feasible for smaller rotorcraft. These are lower-cost HUMS units that can still provide reliable flight recognition data. The incorporation of a global positioning system (GPS) into a HUMS unit, and the benefits of usage monitoring, are also discussed in this section.

- Section 6 proposes possible new guidelines for certification and qualification, the integration of HUMS into the operator's maintenance program, and more rational procedures for obtaining maintenance credits while maintaining structural integrity.

- Section 7 describes the results of the damage tolerance analysis performed on selected components, including the theoretical redesign of four PSEs to meet damage tolerance requirements. The theoretical redesigns, which include both geometric and material changes, compare the impact on weight and inspections relative to baseline safe-life designs.

2. DESCRIPTION OF THE UTILITY MISSION IN MORGAN CITY.

Usage data recorded during the UMMC covered the period from May 12, 1999, through October 4, 1999. A total of 177 hours of data was collected for the UMMC; however, approximately 59 hours of this was ground run data due to various problems PHI was having with the helicopter. Nonetheless, the data sample for the UMMC comprises a reasonable amount of flight time (approximately 118 hours of flight data) compared to the limited ASHM (approximately 17 hours of flight data) and the more comprehensive GCM (approximately 450 hours of flight data). Because of the differences in the amounts of flight time that were collected for each mission, care should be exercised regarding the mission characteristics presented and in any analysis that compare or contrast the three missions.

The UMMC consisted mainly of long level flights at cruise airspeed with some shorter pickup and delivery flights. The broad mission statistics are presented in table 2-1. The mission spectrum detailing the time at condition is tabulated in table 2-2. It should be noted that autorotation is defined, for the purpose of the mission spectrum, as less than 10% combined engine torque while in flight. A comparison between the certification spectrum, GCM spectrum, the ASHM spectrum, and the UMMC is presented in table 2-3. The ASHM consists of a significantly higher percentage time in low to moderate speeds (0.8 and 0.9 V_h) and in turning maneuvers (conditions 34 through 37) than the other spectra. The GCM consisted primarily of high-speed level flight. The UMMC, ASHM, and GCM indicate more time spent at 324 revolutions per minute (rpm) than at 314 rpm, while the certification spectrum assumes more time at 314 rpm. The time at condition comparison is emphasized in figure 2-1, which presents the data sorted by descending time at condition for the UMMC.

TABLE 2-1. UTILITY MISSION IN MORGAN CITY STATISTICS

Period of Mission	5/12/99 Through 10/4/99 Inclusive
Total HUMS recorded hours	77.70 hrs
HUMS recorded flight hours	118.45 hrs
HUMS recorded ground time	59.25 hrs
HUMS recorded flights	272 flights
Average flight duration	26 minutes
Gross weight breakdown	Light 0.04% Medium 5.36% Heavy 94.60%
Altitude breakdown*	<3000 ft 76% 3000-6000 ft 20.63% >6000 ft 3.37%

*When calculating component fatigue lives with an altitude breakdown for the Morgan City Mission, an altitude split was applied between low (≤3000 ft H_d) and high (>3000 ft H_d) altitudes only. Thus, the lives were calculated with an altitude split of 76% (≤3000 ft H_d) and 24% (>3000 ft H_d).

TABLE 2-2. UTILITY MISSION IN MORGAN CITY SPECTRUM

No.	Flight Condition	Percent Time at Gross Weight (lb)			
		<8000	8,000 to 10,000	10,000 to 12,500	Total
1	Rotor Start	0	0	0	0
2	Ground Time (rpm 250-324)	0	0	0	0
3	Normal Shutdown With Collective	0	0	0	0
4	IGE Steady Hover at 314 rpm	0.0042	0.0197	0.3134	0.3373
5	IGE Steady Hover at 324 rpm	0.0166	0.0789	1.2537	1.3492
6	IGE 90° Right Hover Turn	0	0.0365	0.3181	0.3546
7	IGE 90° Left Hover Turn	0	0.0260	0.2417	0.2677
8	IGE Longitudinal Control Reversal	0	0	0.0027	0.0027
9	IGE Lateral Control Reversal	0	0.0028	0.0163	0.0191
10	IGE Rudder Control Reversal	0	0	0.0047	0.0047
11	IGE Right Sideward Flight	0	0.0026	0.0386	0.0412
12	IGE Left Sideward Flight	0	0.0097	0.0589	0.0686
13	IGE Rearward Flight	0	0	0	0
14	Normal Takeoff and Acceleration to Climb Airspeed	0.0222	0.1728	3.5477	3.7427
15	Twin Engine Normal Approach and Landing	0.0009	0.0021	0.0367	0.0398
16	Single Engine Normal Approach and Landing	0	0	0	0
17	0.4 V_h Level Flight at 314 rpm	0	0.0093	0.2328	0.2420
18	0.4 V_h Level Flight at 324 rpm	0	0.0371	0.9310	0.9681
19	0.6 V_h Level Flight at 314 rpm	0	0.0174	0.3176	0.3351
20	0.6 V_h Level Flight at 324 rpm	0	0.0697	1.2705	1.3402
21	0.8 V_h Level Flight at 314 rpm	0	0.0477	0.9690	1.0167
22	0.8 V_h Level Flight at 324 rpm	0	0.1906	3.8760	4.0666
23	0.9 V_h Level Flight at 314 rpm	0	0.1624	2.5462	2.7086
24	0.9 V_h Level Flight at 324 rpm	0	0.6496	10.1848	10.8344
25	1.0 V_h Level Flight at 314 rpm	0	0.6512	11.6491	12.3003
26	1.0 V_h Level Flight at 324 rpm	0	2.6046	46.5965	49.2011
27	V_{ne} at 314 rpm	0	0.0015	0.4494	0.4509
28	V_{ne} at 324 rpm	0	0.0061	1.7976	1.8037
29	Twin Engine Full Power Climb	0	0.2440	3.2057	3.4497
30	Single Engine Full Power Climb	0	0	0	0
31	0.6 V_h Cyclic Pullup	0	0.0022	0.1983	0.2005

TABLE 2-2. UTILITY MISSION IN MORGAN CITY SPECTRUM (Continued)

No.	Flight Condition	Percent Time at Gross Weight (lb)			
		<8000	8,000 to 10,000	10,000 to 12,500	Total
32	0.9 V_h Cyclic Pullup	0	0	0.0926	0.0926
33	Norm. Accel. from Climb A/S - 0.9 V_h	0	0	0	0
34	0.6 V_h Right Turn	0	0.0534	1.1598	1.2132
35	0.9 V_h Right Turn	0	0.0334	0.5795	0.6129
36	0.6 V_h Left Turn	0	0.0429	0.7832	0.8261
37	0.9 V_h Left Turn	0	0.0526	0.5006	0.5532
38	0.9 V_h Longitudinal Control Reversal	0	0	0	0
39	0.9 V_h Lateral Control Reversal	0	0	0	0
40	0.9 V_h Rudder Control Reversal	0	0	0	0
41	Deceleration From 0.9 V_h to Descent A/S	0	0	0	0
42	Twin Engine Partial Power Descent	0	0.1266	1.3916	1.5183
43	Single Engine Partial Power Descent	0	0	0	0
44	Twin to Single Engine in Full Power Climb	0	0	0	0
45	Twin to Single Engine at 0.9 V_h	0	0	0	0
46	Single to Twin Engine in Power Descent	0	0	0	0
47	Twin Engine to Autorotation* at 0.6 V_h	0	0	0.0033	0.0033
48	Twin Engine to Autorotation* at 0.9 V_h	0	0	0.0030	0.0030
49	Stabilized Autorotation* to Twin Engine	0	0	0	0
50	Autorotation* at V_{ne} and Minimum rpm	0	0	0	0
51	Autorotation* at V_{ne} and Maximum rpm	0	0	0	0
52	Autorotation* Right Turn	0	0	0.0118	0.0118
53	Autorotation* Left Turn	0	0.0007	0.0084	0.0091
54	Unrecognized	0	0.0008	0.0102	0.0110
		0.0439	5.3551	94.6011	100.0000

*Autorotation was recorded when combined engine power was less than 10%.

TABLE 2-3. SPECTRA COMPARISON

No.	Certification Spectrum Condition	Certification (%)	Atlanta Short Haul (%)	Gulf Coast (%)	Morgan City (%)
1	Rotor Start[1]	0.5000	0	0	0
2	Ground Time (rpm 250-324)[2]	1.0000	0	0	0
3	Normal Shutdown With Collective[1]	0.5000	0	0	0
4	IGE Steady Hover at 314 rpm	1.0000	1.6022	0.5501	0.3373
5	IGE Steady Hover at 324 rpm	2.0000	3.2529	2.2003	1.3492
6	IGE 90° Right Hover Turn	0.0700	0.9421	0.4330	0.3546
7	IGE 90° Left Hover Turn	0.0700	1.2715	0.3809	0.2677
8	IGE Longitudinal Control Reversal	0.0100	0.0579	0.0331	0.0027
9	IGE Lateral Control Reversal	0.0100	0.0889	0.0359	0.0191
10	IGE Rudder Control Reversal	0.0100	0.0484	0.0968	0.0047
11	IGE Right Sideward Flight	0.2500	0.0151	0.0379	0.0412
12	IGE Left Sideward Flight	0.2500	0.1746	0.0976	0.0686
13	IGE Rearward Flight	0.1000	0	0	0
14	Normal Takeoff and Acceleration to Climb Airspeed	1.5000	6.2583	0.1323	3.7427
15	Twin Engine Normal Approach and Landing	1.4300	0.1262	0.5461	0.0398
16	Single Engine Normal Approach and Landing	0.0300	0	0.0084	0
17	0.4 V_h Level Flight at 314 rpm[3]	0.8000	0.6312	0	0.2420
18	0.4 V_h Level Flight at 324 rpm[3]	0.2000	2.5246	0	0.9681
19	0.6 V_h Level Flight at 314 rpm	2.4000	1.1091	0.4379	0.3351
20	0.6 V_h Level Flight at 324 rpm	0.6000	4.4365	1.7514	1.3402
21	0.8 V_h Level Flight at 314 rpm	12.0000	6.4399	0.6736	1.0167
22	0.8 V_h Level Flight at 324 rpm	3.0000	25.7597	2.6945	4.0666
23	0.9 V_h Level Flight at 314 rpm	16.0000	3.4167	2.2297	2.7086
24	0.9 V_h Level Flight at 324 rpm	4.0000	13.6669	8.9187	10.8344
25	1.0 V_h Level Flight at 314 rpm	30.4000	1.2926	12.6411	12.3003
26	1.0 V_h Level Flight at 324 rpm	7.6000	5.1705	50.5644	49.2011
27	V_{ne} at 314 rpm	0.8000	0	0.4511	0.4509
28	V_{ne} at 324 rpm	0.2000	0	1.8046	1.8037
29	Twin Engine Full Power Climb	4.7500	2.8391	6.3150	3.4497
30	Single Engine Full Power Climb	0.1200	0	0.0013	0

TABLE 2-3. SPECTRA COMPARISON (Continued)

No.	Certification Spectrum Condition	Certification (%)	Atlanta Short Haul (%)	Gulf Coast (%)	Morgan City (%)
31	0.6 V_h Cyclic Pullup	0.1500	0.6651	0.0862	0.2005
32	0.9 V_h Cyclic Pullup	0.0500	0.0294	0.0182	0.0926
33	Norm. Accel. From Climb A/S - 0.9 V_h	1.0000	0	0	0
34	0.6 V_h Right Turn	1.0000	4.7646	1.2422	1.2132
35	0.9 V_h Right Turn	1.0000	4.0439	0.2726	0.6129
36	0.6 V_h Left Turn	1.0000	2.8248	0.4894	0.8261
37	0.9 V_h Left Turn	1.0000	4.3725	0.3962	0.5532
38	0.9 V_h Longitudinal Control Reversal	0.0500	0	0	0
39	0.9 V_h Lateral Control Reversal	0.0500	0	0	0
40	0.9 V_h Rudder Control Reversal	0.0500	0	0	0
41	Deceleration From 0.9 V_h to Descent A/S	0.1800	0	0	0
42	Twin Engine Partial Power Descent	2.6440	2.0914	4.1055	1.5183
43	Single Engine Partial Power Descent	0.1300	0	0.0323	0
44	Twin to Single Engine in Full Power Climb	0.0100	0	0.0003	0
45	Twin to Single Engine at 0.9 V_h	0.0100	0	0.0065	0
46	Single to Twin Engine in Power Descent	0.0100	0	0.0051	0
47	Twin Engine to Autorotation[4] at 0.6 V_h	0.0050	0.0032	0.0003	0.0033
48	Twin Engine to Autorotation[4] at 0.9 V_h	0.0050	0.0032	0.0001	0.0030
49	Stabilized Autorotation[4] to Twin Engine	0.0100	0	0	0
50	Autorotation[4] at V_{ne} and Minimum rpm	0.0200	0	0	0
51	Autorotation[4] at V_{ne} and Maximum rpm	0.0200	0	0	0
52	Autorotation[4] Right Turn	0.0030	0.0349	0.0128	0.0118
53	Autorotation[4] Left Turn	0.0030	0	0.0071	0.0091
54	Unrecognized	0	0.0421	0.2895[5]	0.0110
	Total	100.0000	100.0000	100.0000	100.0000

(1) Rotor starts and shutdowns were considered as events. The main rotor yoke was the only affected component out of the four selected components.

(2) Ground time was added after spectrum analysis, therefore it is excluded from the spectrum.

(3) 0.4V_h data were missing from the gulf coast data.

(4) Autorotation was recorded when the combined engine power was less than 10%

(5) Unrecognized data were reduced to 0.05% for component fatigue life calculations.

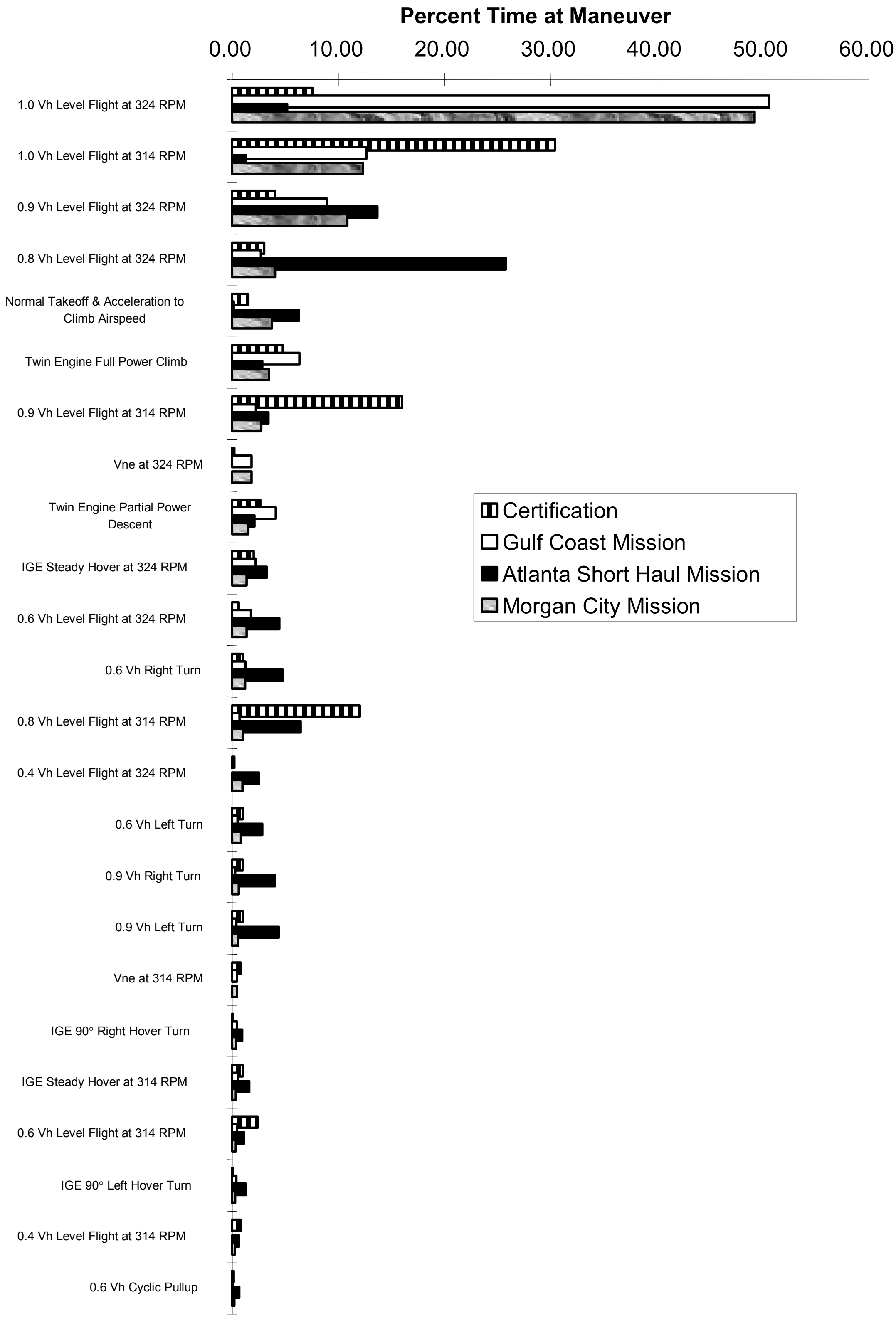

FIGURE 2-1. SPECTRA COMPARISON HISTOGRAMS

The gross weight (GW) for the UMMC mission was measured as 0.04% of the time at less than 8,000 lbs, 5.36% of the time between 8,000 and 10,000 lbs, and 94.6% of the time at greater than 10,000 lbs. This is shown in table 2-1. During this study, the pilots were asked to enter the GW into the HUMS unit before the flight. However, if this was not done, the GW defaulted to 12,000 lbs. Because of the large percentage time above 10,000 lbs (94.6%), it is suspected that the GW was not entered routinely during the UMMC. The HUMS unit was installed and removed several times during the course of the UMMC study, which may have contributed to the lack of consistency in entering the GW. This potentially could be an issue with commercial HUMS units installed into a fleet of helicopters; however, this problem could be overcome with adequate training and with a more reliable and available HUMS unit, or possibly with a HUMS unit that measures and records the GW automatically. For the purposes of this report, the GW breakdown as stated above was used.

Ground-running time is not included in the time-at-condition spectrum but is calculated separately so that damage can be related to flight time. The certification process also assumes the time spent in ground-running and, similarly, sums that damage into the 100-hour flight spectrum damage before calculating a life.

3. DESCRIPTION OF SELECTED PRINCIPAL STRUCTURAL ELEMENTS.

The four PSEs components selected for analysis, which are part of the hub and blade assembly of the 412 helicopter shown in figure 3-1, are as follows.

- Rephase lever
- Collective lever
- Main rotor spindle
- Main rotor yoke

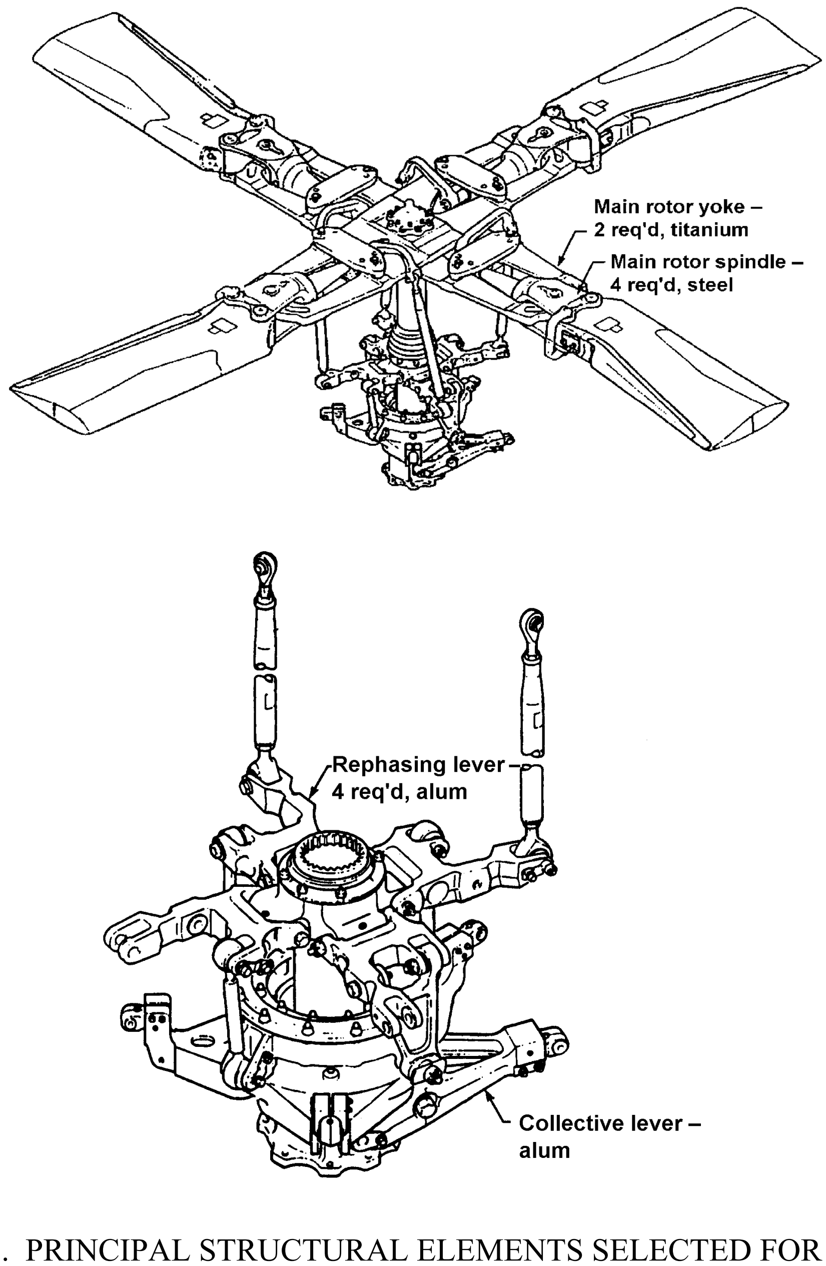

FIGURE 3-1. PRINCIPAL STRUCTURAL ELEMENTS SELECTED FOR 412 HUMS DAMAGE TOLERANCE STUDY

The rephase lever (see figure 3-2) is manufactured from a 7075-T73 aluminum forging. The rephase lever pivots on a rotating hub and provides a reindexing of the pitch link to the swashplate by offsetting the attach points. Swashplate motion is imparted to the rephase lever via a tubular link or a drive link. This motion is then transferred to the rotor by the pitch link with the rephase lever as the intermediate mechanism. The rephase lever has a retirement life of 5000 hours. The collective lever (see figure 3-3) is manufactured from a 7075-T73 aluminum forging. The collective boost actuator attaches at the apex of the lever. The lever pivots about an axis common to a lug situated on the swashplate support. The ends of the legs attach to the collective sleeve to impart mean blade angle changes. The collective lever has a retirement life of 10,000 hours.

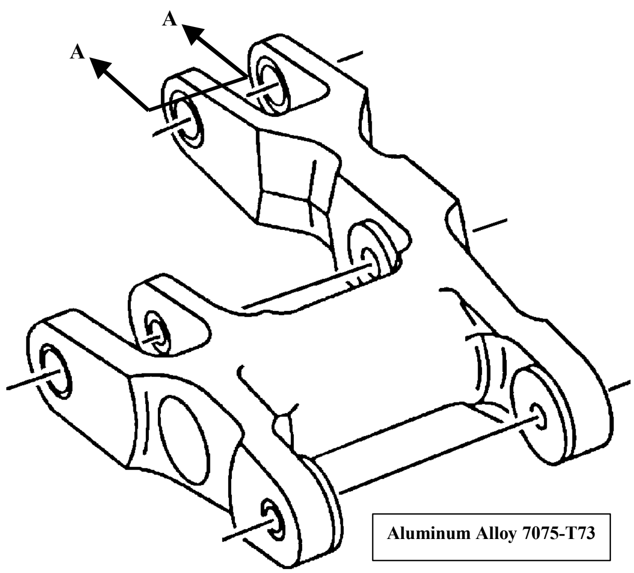

FIGURE 3-2. REPHASE LEVER GEOMETRY

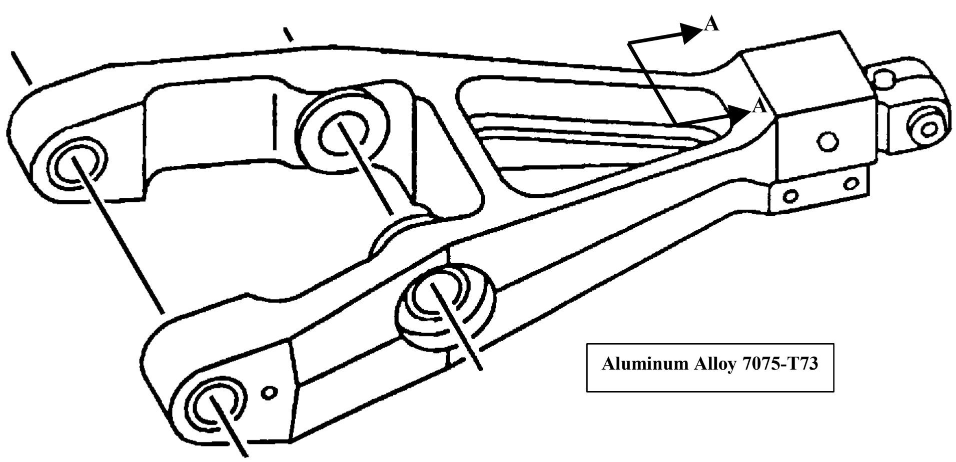

FIGURE 3-3. COLLECTIVE LEVER GEOMETRY

The main rotor spindle (see figure 3-4) attaches the main rotor blade to the hub and imparts cyclic and collective pitching motion to the main rotor blade. The original spindle design was manufactured from SAE 4340 alloy steel and was protected from corrosion by an applied surface finish. The elastomeric feathering bearing was mechanically attached to the spindle by means of a bonded inner race. The pitch horn is splined to the end of the spindle. The retirement life of the 4340 alloy steel spindle is 5000 hours. Later designs of the spindle are made from 15-5 PH stainless steel to eliminate corrosion problems. The elastomeric feathering bearing in the later design is molded directly to the spindle surface, allowing the elastomeric element to be increased in size, thereby, reducing strains. The retirement life for the 15-5 PH stainless steel spindle is 10,000 hours. In this report, the analysis assumed that the 15-5 PH stainless steel spindle was installed.

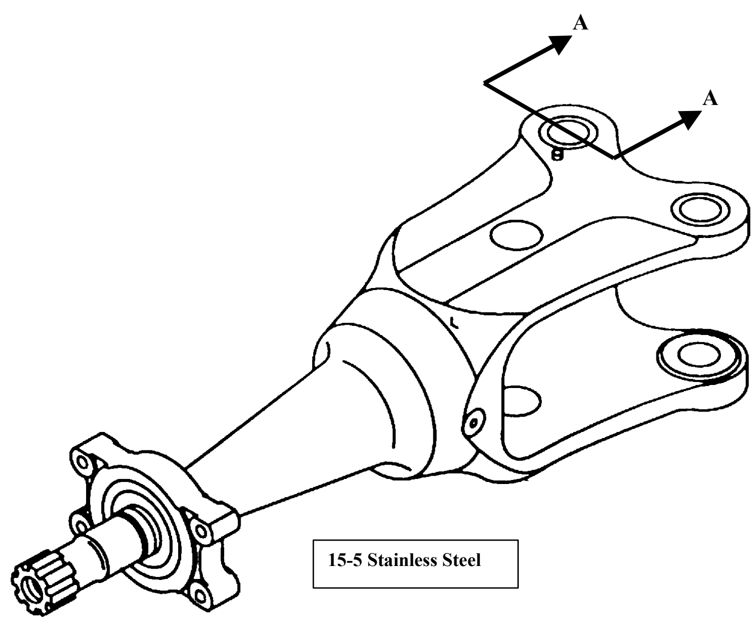

FIGURE 3-4. MAIN ROTOR SPINDLE GEOMETRY

The main rotor yoke (see figure 3-5) attaches the hub to the main rotor mast and allows for beamwise flapping of the main rotor blade. The original design of the main rotor yoke was initially certified with a 5000-hour life. In two separate incidents, the yoke sustained a partial flexure fatigue crack (noncatastrophic) after ground static compressive overloads due to high surface winds. The high loads compressively yielded the shotpeened surface of the 6Al-4V-annealed titanium flexure, nullifying the benefits of the shotpeening. A 700-hour service life was established for these early yokes by the manufacturer’s bulletin and an FAA Airworthiness Directive. The yoke was redesigned to solve this problem. The yoke flexure was lengthened, the material changed to 6Al-4V beta solution heat stress treat and overage (BSHTOA), and a dynamically activated droop stop was incorporated to protect the yoke flexure against high beamwise loads due to natural winds or winds generated by other helicopters operating nearby when the rotor was not operating. The retirement life for the redesigned yokes with longer flexures and droop stops is 5000 hours. In this report, the analysis is done assuming these newer yokes are installed on the helicopter.

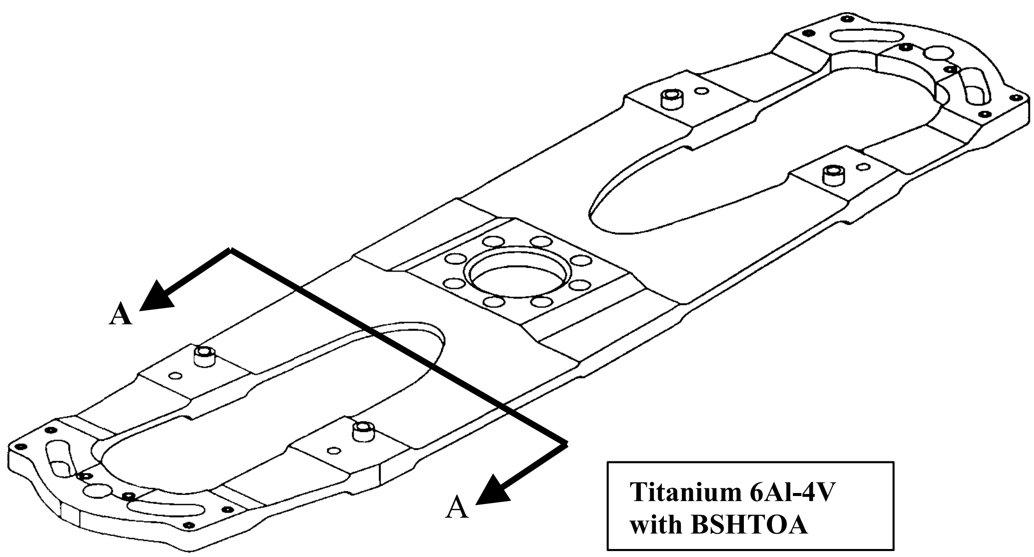

FIGURE 3-5. MAIN ROTOR YOKE GEOMETRY

4. USAGE MONITORING AND FATIGUE LIFE ANALYSIS.

4.1 ANALYSIS PROCEDURE.

A fatigue analysis of the UMMC usage data was performed on a basis that is consistent with the certification of the selected PSEs. Figure 4-1 shows a simplified overview of the analysis procedure. Note that the fatigue life methodology using a measured usage spectrum from HUMS remains unchanged from that used in the certification process with an estimated usage spectrum. The only variation in assumptions from the certification procedure is the use of measured usage time-at-condition in place of the estimated usage. In addition to the certification procedure, component lives were calculated that include altitude effects.

- Time-at-condition is determined from analysis of the measured flight parameters using flight condition recognition (FCR) software (see references 1 and 2 for FCR description).

- The loads for each condition are taken from the certification load survey. No additional measured loads are used in the HUMS data processing.

- Component damage is calculated by combining the loads with the time-at-condition using the certification endurance limits.

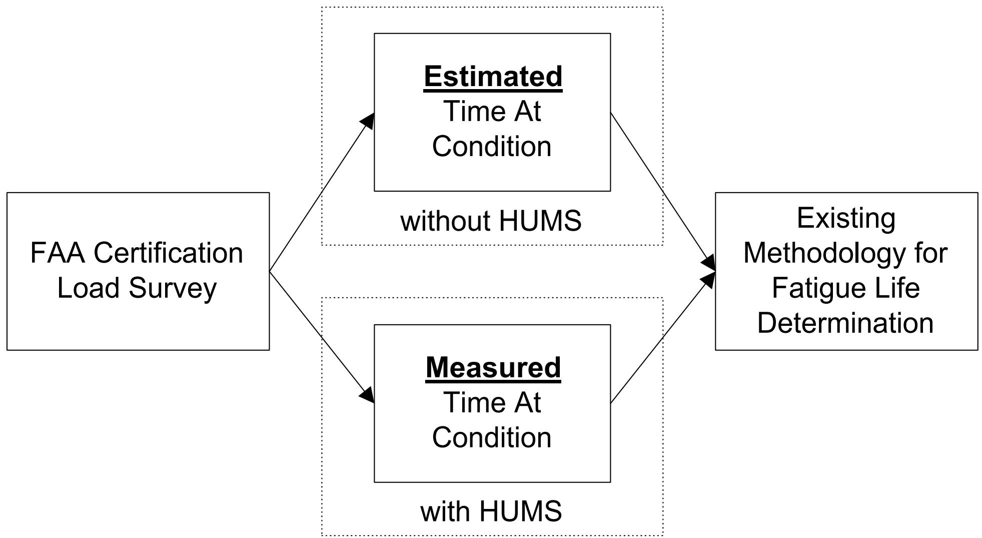

FIGURE 4-1. FATIGUE LIFE METHODOLOGY WITH AND WITHOUT HUMS

The certification methodology uses an assumed worst-case spectrum of time-at-condition to determine the life of helicopter components. When the FCR software processes recorded data, there is a small percentage of flight time that is not within the parameter set associated with any of the defined conditions. This time is considered to be unrecognized and is assigned the most damaging condition within the domain in which the event occurred.

The FCR software used in reference 2 to process the GCM data was enhanced to reduce the time in unrecognized flight conditions. It was observed that the percentage of unrecognized condition reduced significantly when the ASHM data were processed though the revised FCR software. Reprocessing of the 450 hours of the GCM data was beyond the scope of the current effort. Instead, the assumption was made that the software enhancements would have reduced the percentage of unrecognized maneuvers to an amount similar to that seen for the ASHM. Consequently, by redistributing the excess unrecognized time in the proportion of the recorded spectrum, the percentage of unrecognized condition was reduced for the GCM to approximately the level seen in the ASHM data. The lives were recomputed on that basis. The contribution of unrecognized conditions to total damage is indicated in tables 2-2 and 2-3.

As shown in figure 4-2, a potential benefit from usage monitoring is part retirement extension if the actual usage severity is milder than the basis for certification. However, recommended retirement lives derived for HUMS-equipped aircraft may be subject to limiting factors other than fatigue calculations. For example, maximum lives or minimum usage rates may be restricted due to reasons of practicality, including but not limited to, corrosion, wear, and component sensitivity to load variation.

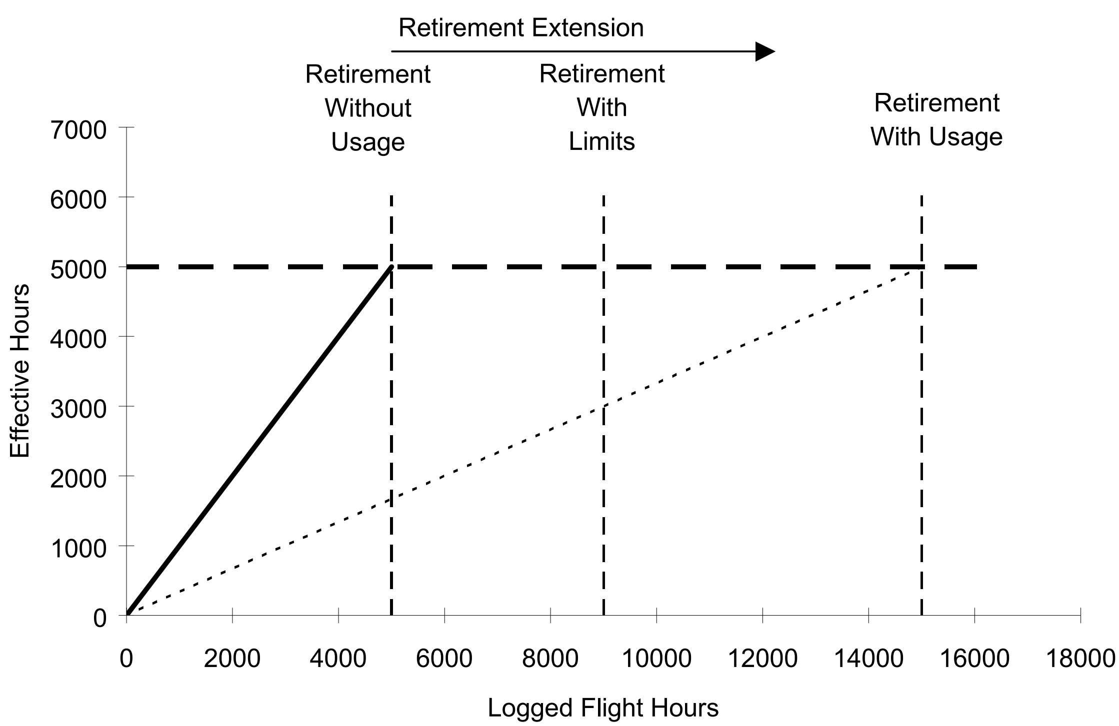

FIGURE 4-2. EFFECT ON RETIREMENT OF USAGE MONITORING

4.2 FATIGUE LIFE COMPARISONS OF PSEs FOR VARIOUS USAGE SPECTRA.

Analysis results comparing fatigue safe lives of the PSEs for the UMMC, ASHM, GCM, and certification mission spectrum are summarized in figures 4-3 through 4-6 and in tables 4-1 through 4-4. The rate at which life is being consumed relative to certification is referred to as the component clock rate. If usage indicates that the part is using life faster than certification (i.e., it has a reduced life), then the part is said to have a fast clock. The component safe lives were calculated without regard to altitude for direct comparison to the certification data because certification does not employ an altitude breakdown. Components are certified using the most severe altitude within any condition. However, in this study, pressure altitude (H_p) and outside air temperature are recorded by the HUMS, allowing for the calculation of density altitude (H_d), which is required to take credit for altitude.

Load-level survey data, used as the basis for all life calculations, does not contain all data at all altitudes. For each condition, the survey contains records at 3,000 ft and records at 6,000 ft and/or 12,000 ft for each of the GW, center of gravity combinations flown. Therefore, safe lives were also calculated using a split between high (>3000 ft H_d) and low (≤3000 ft H_d) altitude data to ensure multiple records from which to select the most severe condition. This approach deviates from results previously published for the GCM data [2], which employed a full altitude breakdown. Calculations performed without an altitude split compare directly with certification data. Comparison of spectra with, and without, an altitude split reveals that there are additional potential benefits due to HUMS.

The conclusions that can be drawn from the comparison of the UMMC, ASHM, and GCM fatigue lives to the certification mission are as follows:

- Rephase Lever—With no altitude split, the GCM-calculated life is higher, the UMMC life is about the same and ASHM is lower than the certification fatigue life. With an altitude split, all HUMS mission fatigue lives are much higher than the certification fatigue life. (Note that the HUMS both with and without an altitude split are presented).

- Collective Lever—With no altitude split, the UMMC, GCM, and ASHM lives were 23% to 42% higher than certification and much higher with an altitude split.

- Main Rotor Spindle—With no altitude split, the GCM life is higher, and the UMMC and ASHM lives are lower, than certification. All HUMS missions are higher with an altitude split.

- Main Rotor Yoke—With no altitude split, the UMMC and GCM lives are higher, and the ASHM life is lower, than certification. With an altitude split, the UMMC and GCM are higher than the certification mission, and the ASHM is about the same as certification.

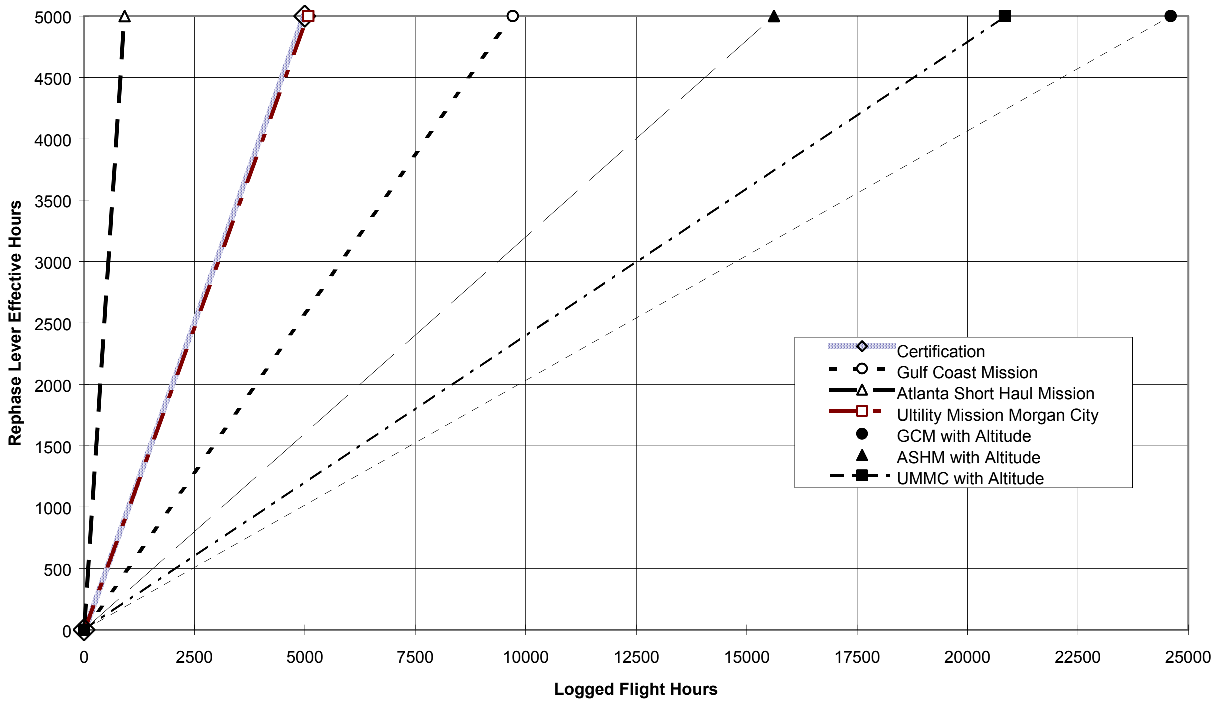

FIGURE 4-3. EFFECTIVE USAGE—REPHASE LEVER

TABLE 4-1. REPHASE LEVER CALCULATED FATIGUE LIFE

		Calculated Hours	% of Certification	Clock Rate[1,2]	URD[3] (%)
No Altitude Split	Certification Mission	5,000	100	100%	0
	Gulf Coast Mission	9,710	194	51%	8
	Atlanta Short Haul Mission	920	18	543%	1
	Morgan City Utility Mission	5,080	102	98%	1
Low/High Altitude	Gulf Coast Mission	24,610	492	20%	8
	Atlanta Short Haul Mission	15,620	312	32%	1
	Morgan City Utility Mission	20,850	417	24%	1

(1) Clock Rate—the rate of life consumption relative to certification.
(2) Limitations (see section 4.2) may apply that restrict usage clock rate.
(3) Unrecognized damage percentage (URD) %—Damage contribution from unrecognized conditions.

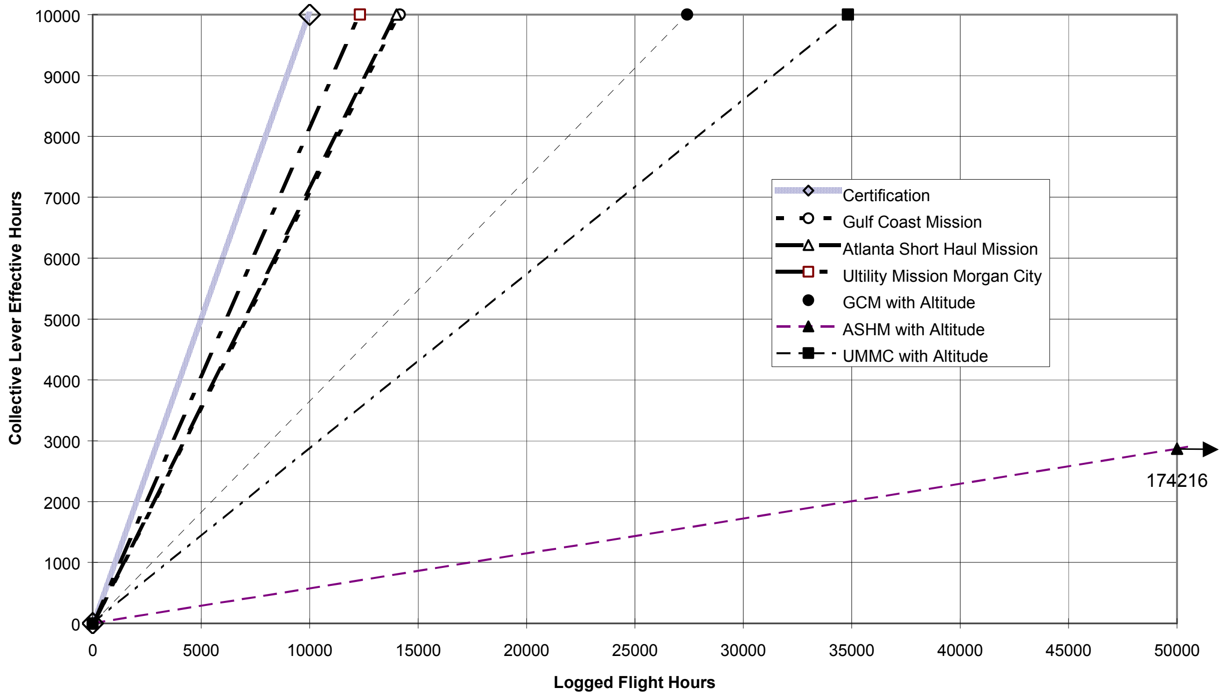

FIGURE 4-4. EFFECTIVE USAGE—COLLECTIVE LEVER

TABLE 4-2. COLLECTIVE LEVER CALCULATED FATIGUE LIFE

		Calculated Hours	% of Certification	Clock Rate[1,2]	URD[3] (%)
No Altitude Split	Certification Mission	10,000	100	100%	0
	Gulf Coast Mission	14,160	142	71%	7
	Atlanta Short Haul Mission	14,010	140	71%	5
	Morgan City Utility Mission	12,330	123	81%	2
Low/High Altitude	Gulf Coast Mission	27,410	274	36%	6
	Atlanta Short Haul Mission	174,220	1742	6%	8
	Morgan City Utility Mission	34,830	348	29%	2

(1) Clock Rate—the rate of life consumption relative to certification.
(2) Limitations (see section 4.2) may apply that restrict usage clock rate.
(3) URD %—damage contribution from unrecognized conditions.

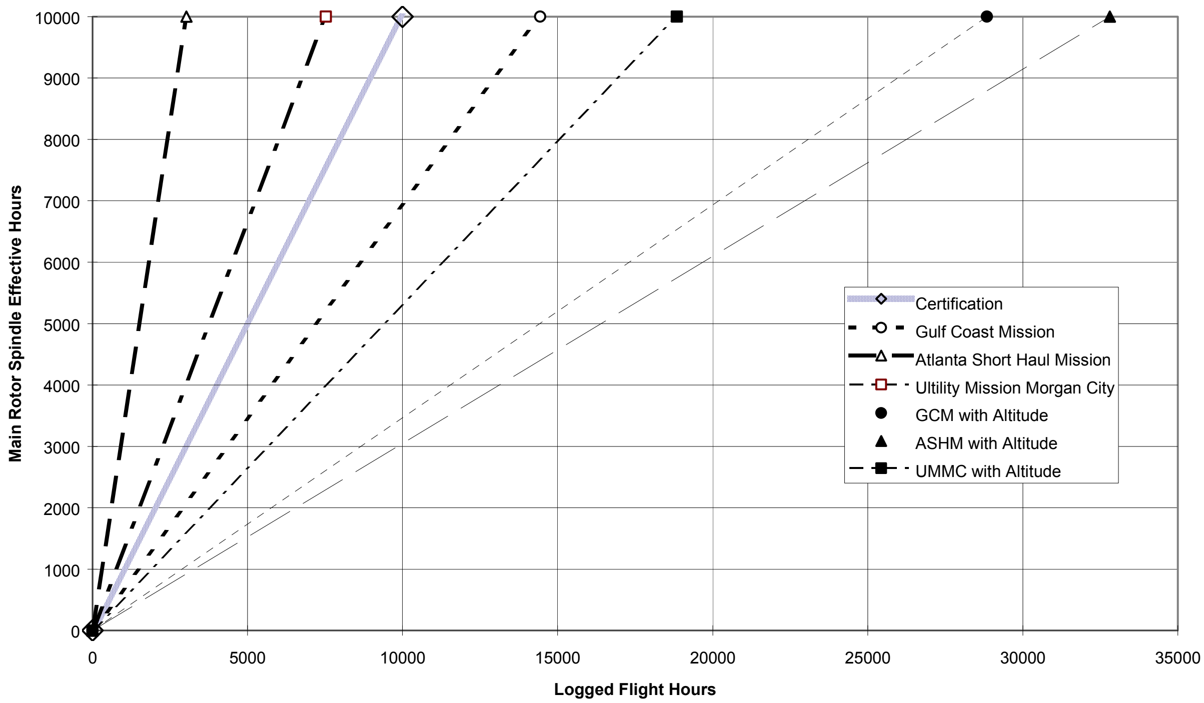

FIGURE 4-5. EFFECTIVE USAGE—MAIN ROTOR SPINDLE

TABLE 4-3. MAIN ROTOR SPINDLE CALCULATED FATIGUE LIFE

		Calculated Hours	% of Certification	Clock Rate[1,2]	URD[3] %
No Altitude Split	Certification Mission	10,000	100	100%	0
	Gulf Coast Mission	14,440	144	69%	11
	Atlanta Short Haul Mission	3,030	30	330%	2
	Morgan City Utility Mission	7,530	75	133%	2
Low/High Altitude	Gulf Coast Mission	28,840	288	35%	18
	Atlanta Short Haul Mission	32,810	328	30%	16
	Morgan City Utility Mission	18,850	188	53%	5

(1) Clock Rate—the rate of life consumption relative to certification.
(2) Limitations (see section 4.2) may apply that restrict usage clock rate.
(3) URD %—damage contribution from unrecognized conditions.

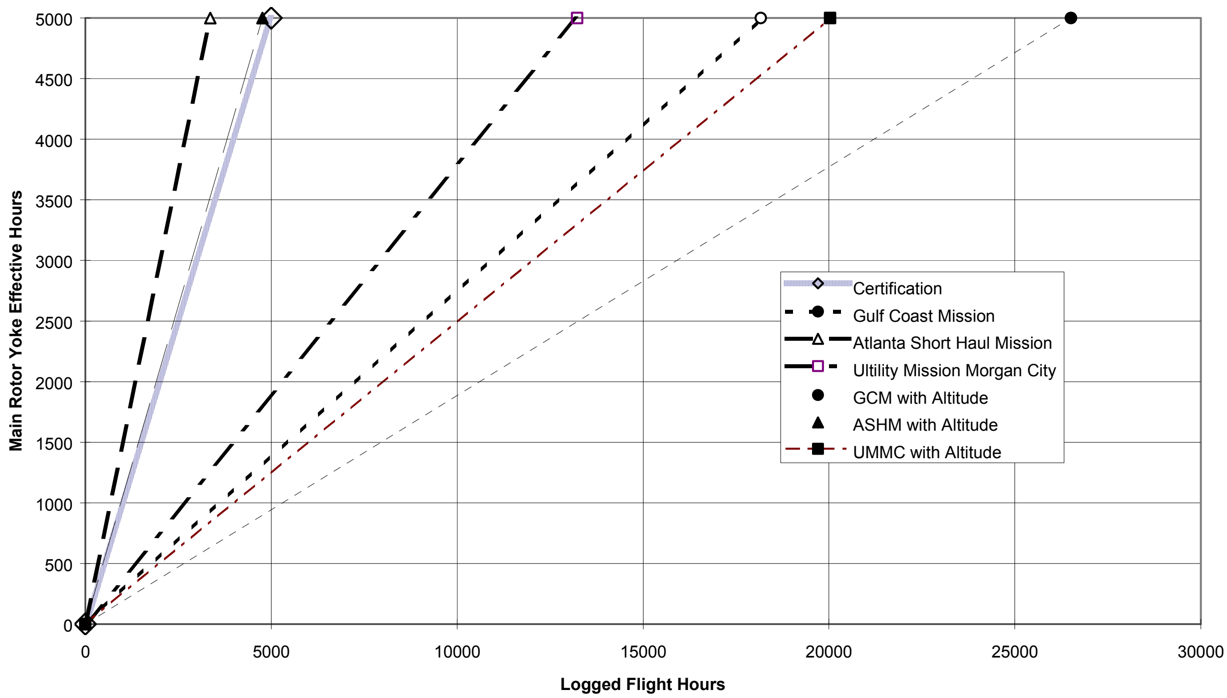

FIGURE 4-6. EFFECTIVE USAGE—MAIN ROTOR YOKE

TABLE 4-4. MAIN ROTOR YOKE CALCULATED FATIGUE LIFE

		Calculated Hours	% of Certification	Clock Rate[1,2]	URD[3] %
No Altitude Split	Certification Mission	5000	100	100%	0
	Gulf Coast Mission	18,170	363	28%	11
	Atlanta Short Haul Mission	3360	67	149%	2
	Morgan City Utility Mission	13,230	265	38%	1
Low/High Altitude	Gulf Coast Mission	26,510	530	19%	10
	Atlanta Short Haul Mission	4760	95	105%	3
	Morgan City Utility Mission	20,030	400	25%	1

(1) Clock Rate—the rate of life consumption relative to certification.
(2) Limitations (see section 4.2) may apply that restrict usage clock rate.
(3) URD %—damage contribution from unrecognized conditions.

4.3 MEASURED LOAD COMPARISON.

This section includes comparisons of measured loads data versus derived loads data from usage monitoring for the ASHM, GCM, and certification mission types. A very limited set of oscillatory loads data was measured during the ASHM. These data include the collective boost tube, left cyclic boost tube, and right cyclic boost tube. These loads data were collected to compare to loads measured during the 412 load-level survey.

This comparison of loads was used to indicate the amount of conservatism that is built into a standard fatigue analysis using load-level survey data. These data were analyzed to determine the frequency of occurrence at various load levels and were processed to generate the measured load exceedance curves presented in figures 4-7 through 4-9. The curve represents the number of times per hour a given oscillatory load exceeded a specified load threshold, e.g., 47 cycles/hour exceeded 200 lb for the collective boost tube (figure 4-7).

Recorded loads data were also extracted from the load-level survey database and were processed with the time-at-condition measured for the ASHM, GCM, and certification missions. These data were then processed as above and plotted for comparison in figures 4-7 through 4-9. The measured ASHM boost tube loads were compared with the boost tube loads measured during the load-level survey using the certification spectrum, the ASHM spectrum, and the GCM spectrum, assuming no altitude or GW breakdown with these spectrums. This comparison shows that the number of load cycles per flight hour at or above the endurance limit in the load-level survey was approximately 100 times more than during the ASHM. The left and right boost tube plots exhibit similar characteristics.

Also included in figures 4-7 through 4-9 are the certification loads using the ASHM mission spectrum with the application of the ASHM GW and altitude breakdowns and an assumed equal split between the forward and aft center of gravity. In the collective boost tube load plot, an additional curve was added showing the certification loads with the ASHM mission spectrum, and the ASHM GW and altitude breakdowns, but without a center of gravity breakdown. As would be expected, these curves are closer to the exceedance curves for the measured ASHM loads, and for some cases, at the low and medium part of the load spectrum, they actually fall below the measured ASHM loads.

At the upper end of the load spectrum, these exceedance curves still lie above the ASHM measured loads, indicating that for loads near or above the endurance limits, the HUMS methodology using the certification loads is still conservative even with an altitude, GW, and center of gravity breakdown, at least in this particular case. These curves do suggest, however, that the finer the HUMS spectrum is sliced, the closer one gets to the actual loads being experienced by the parts on the helicopter. This indicates that it may be advantageous not to slice the HUMS spectrum data too fine in order to have a comfortable level of conservatism in the processing of HUMS spectrum data.

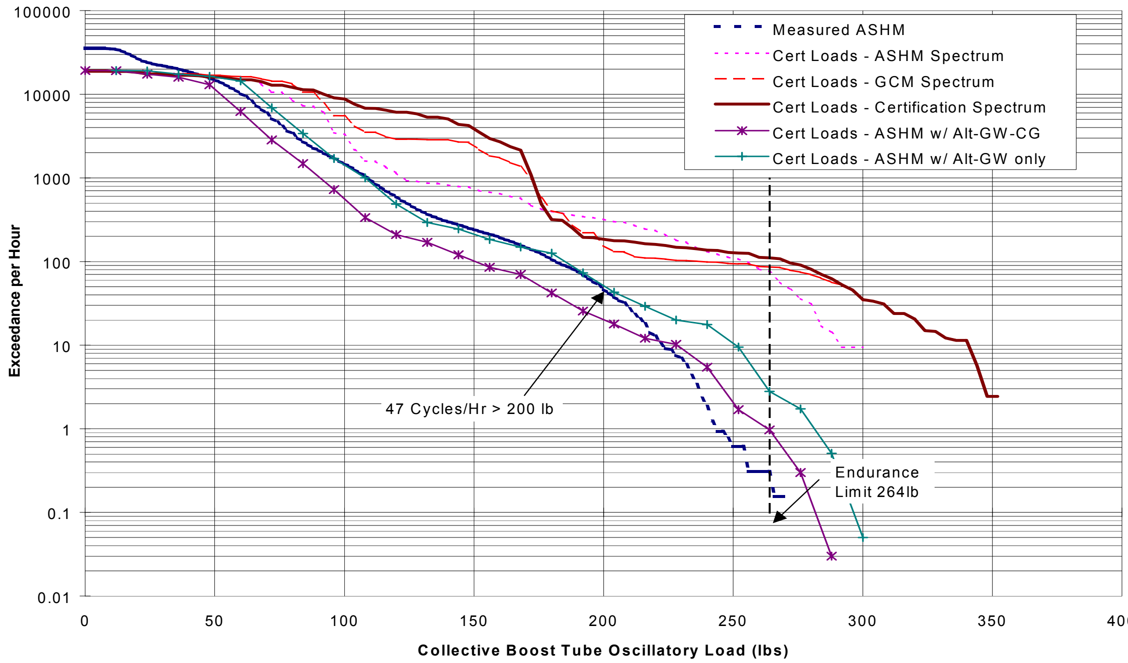

FIGURE 4-7. COLLECTIVE BOOST TUBE LOAD COMPARISON

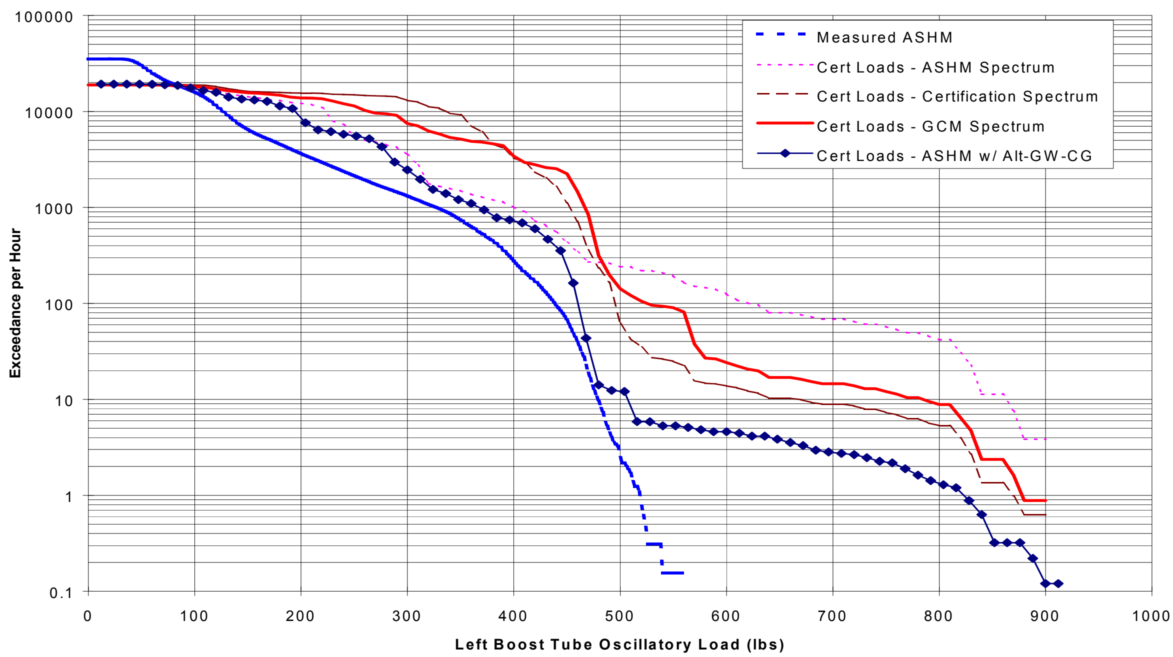

FIGURE 4-8. LEFT BOOST TUBE LOAD COMPARISON

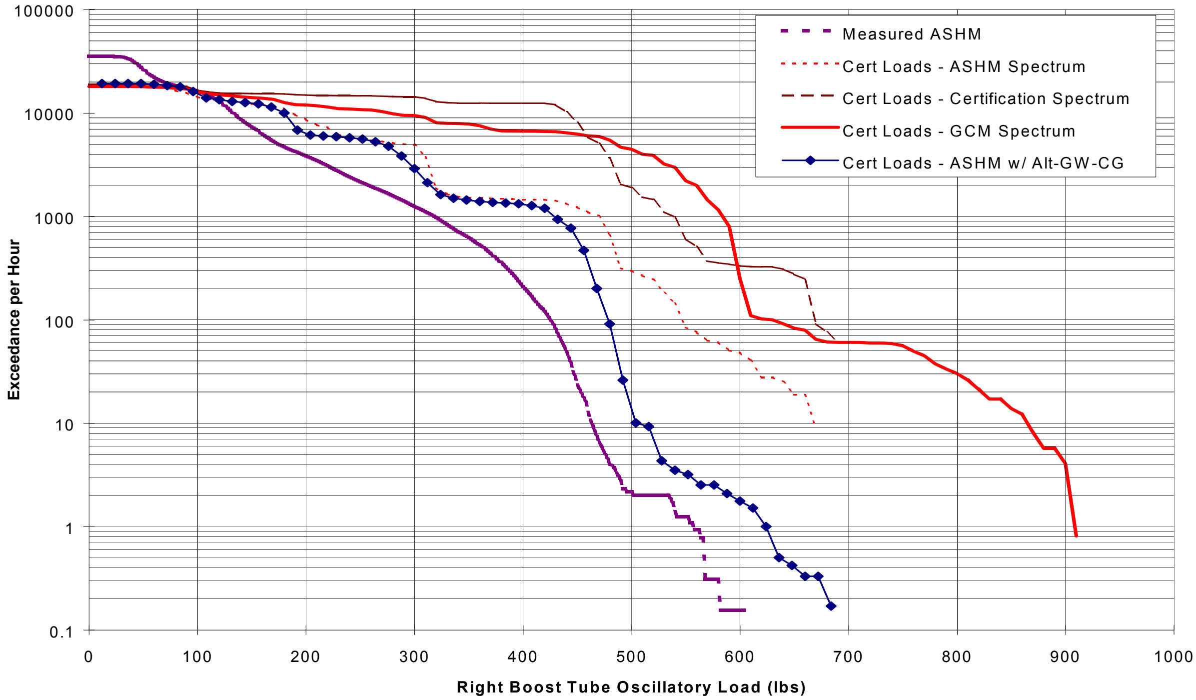

FIGURE 4-9. RIGHT BOOST TUBE LOAD COMPARISON

5. MINI-HUMS INVESTIGATION.

5.1 MINI-HUMS DESCRIPTION.

To reduce the cost of HUMS in the future while obtaining maximum benefit, three simplified or mini-HUMS concepts were investigated. The three simplified concepts reduce the number of sensors and, therefore, reduce the complexity and the cost of the HUMS by only recognizing selected conditions and parameters (e.g., altitude, normal acceleration, airspeed, vertical velocity, and roll angle). The three mini-HUMS concepts are shown in more detail in figure 5-1. It should be noted that with all of these concepts, just as with a complete HUMS, the HUMS unit must determine if the aircraft is airborne or is on the ground. This can be accomplished by using an algorithm that uses rotor torque, rotor rpm, and vertical velocity to determine if the aircraft is airborne. All three of these parameters are measured in a standard helicopter avionics package. Thus, it would simply be a matter of the HUMS unit tapping into and recording these existing measurements.

5.1.1 Mini-HUMS Concept One.

This concept simply records altitude and applies an altitude breakdown to the certification spectrum. Thus, the HUMS unit is essentially a recording altimeter. This is equivalent to producing two certification data sets, one for below 3000 ft and another for at or above 3000 ft. The fatigue life calculations were reprocessed with the above altitude assumptions for all three mini-HUMS proposals, and the results are presented in section 5.2.

5.1.2 Mini-HUMS Concept Two.

This concept records altitude, normal acceleration, airspeed, and vertical velocity. Using these parameters, it can be determined if the aircraft is in level flight and the actual time in level flight calculated. This percentage of time in level flight at various airspeeds is compared to the time in level flight at various airspeeds for the certification mission. The rest of the percentage time from the certification mission is factored accordingly to account for the difference in level flight time between the certification mission and the actual mission as recorded by the HUMS unit. This factored certification spectrum would then be used for the percentage time for all conditions other than level flight. This proposal assumes the helicopter is flown to the certification GW breakdown. As with proposal one above, the actual altitude breakdown recorded by the HUMS unit is used.

5.1.3 Mini-HUMS Concept Three.

This concept records altitude, normal acceleration, airspeed, vertical velocity, and roll angle. Using these parameters, it can be determined if the aircraft is in level flight, is turning, or is performing a pullup, with the actual time in these conditions being calculated. The percentages of time in level flight, turns, and pullups at various airspeeds are compared to the corresponding times for the certification mission.

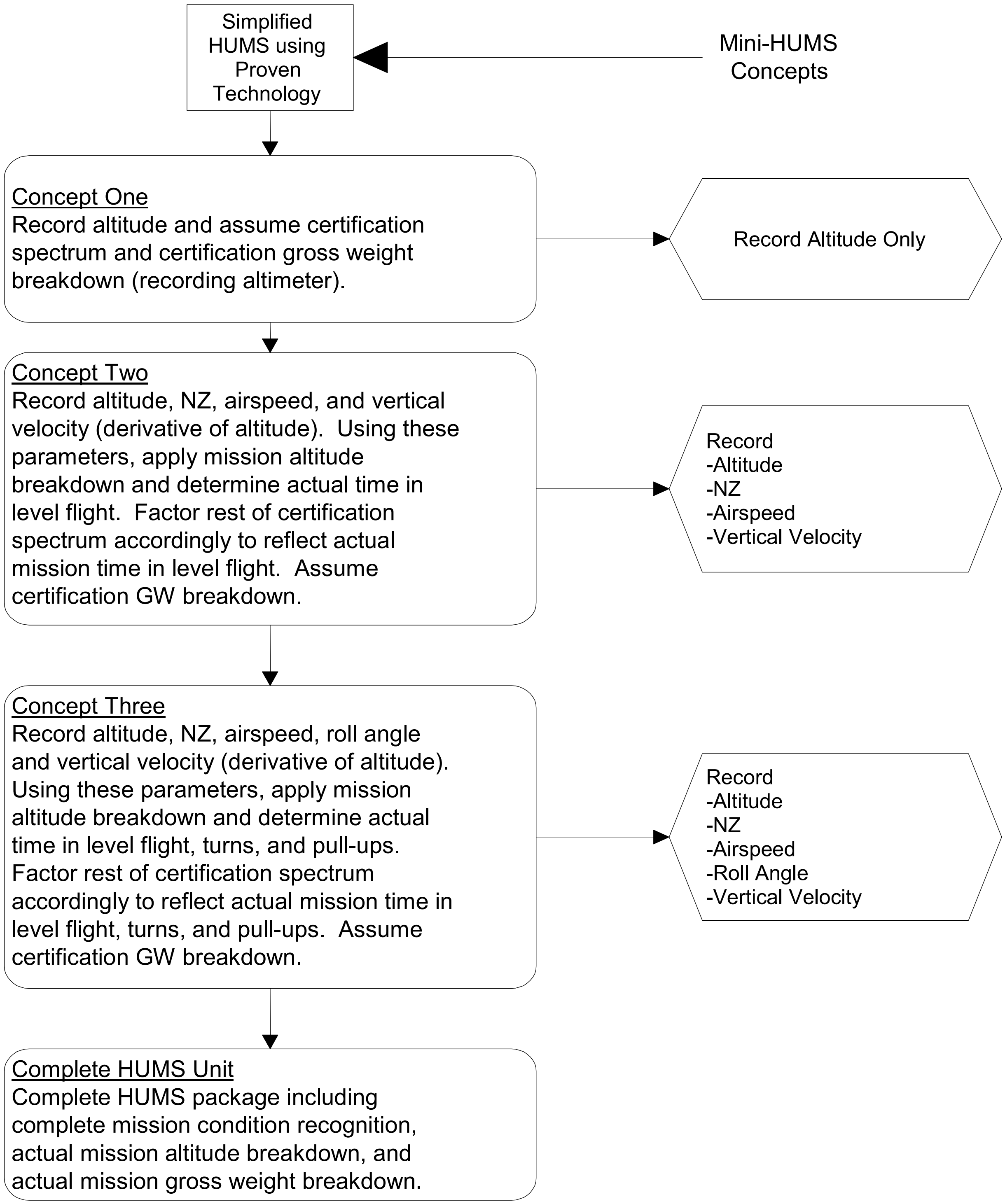

FIGURE 5-1. MINI-HUMS CONCEPTS VERSUS COMPLETE HUMS UNIT

The remaining percentage time from the certification mission is then factored to account for the differences in level flight, turn, and pullup times between the certification mission and the actual mission, as recorded by the HUMS unit. This factored certification spectrum would then be used for the percentage time for all conditions other than level flight, turns, and pullups. This concept assumes the helicopter is flown to the certification GW breakdown. As with concepts one and two, the actual altitude breakdown recorded by the HUMS unit is used.

5.2 SUMMARY OF MINI-HUMS AND COMPLETE HUMS FATIGUE LIVES.

In table 5-1, the fatigue lives for the selected PSEs are shown. Shown first are the currently recommended fatigue lives with no altitude breakdown, then the fatigue lives using the three different mini-HUMS alternatives along with fatigue lives using the complete HUMS package.

TABLE 5-1. SUMMARY OF MINI-HUMS AND COMPLETE HUMS FATIGUE LIVES USING THE MISSION ALTITUDE BREAKDOWN

Life Calculation Method	REPHASE LEVER Fatigue Life (Hrs)			COLLECTIVE LEVER Fatigue Life (Hrs)			MAIN ROTOR SPINDLE Fatigue Life (Hrs)			MAIN ROTOR YOKE Fatigue Life (Hrs)		
- Certification Spectrum - No Altitude Breakdown - Certification Gross Weight Breakdown	**5000**			**10,000**			**10,000**			**5,000**		
Concept \ HUMS Mission Profile	GCM	ASHM	UMMC	GCM	ASHM	UMMC	GCM	ASHM	UMMC	GCM	ASHM	UMMC
Mini HUMS Concept 1 - Certification Spectrum - Mission Altitude Breakdown - Certification Gross Weight Breakdown	**12,910**	**80,320**	**21,030**	**20,730**	**45,170**	**27,607**	**19,000**	**33,090**	**23,563**	**5,760**	**5,460**	**5275**
Mini HUMS Concept 2 - Certification Spectrum but with actual Level Flight from Mission Spectrum - Mission Altitude Breakdown - Certification Gross Weight Breakdown	**16,176**	**53,604**	**31,547**	**30,972**	**56,341**	**45,097**	**23,445**	**20,480**	**35,185**	**11,045**	**5,250**	**11,826**
Mini HUMS Concept 3 - Certification Spectrum but with actual Level Flight, Turns & Pullups from Mission Spectrum - Mission Altitude Breakdown - Certification Gross Weight Breakdown	**40,592**	**21,031**	**34,799**	**32,415**	**72,364**	**46,654**	**33,103**	**22,678**	**32,506**	**9,814**	**3,735**	**9486**
Complete HUMS Package - Mission Spectrum (including unrecognized) - Mission Altitude Breakdown - Mission Gross Weight Breakdown	**24,610**	**15,620**	**20,850**	**27,410**	**174,220**	**34,830**	**28,840**	**32,810**	**18,850**	**26,510**	**4,760**	**20,030**

As already mentioned, the three mini-HUMS all use the mission altitude breakdown, as does the complete HUMS package. Compared to the currently recommended fatigue lives with no altitude breakdown using the certification spectrum, the fatigue lives using the mini-HUMS proposals are, in general, significantly greater. The only part in which this is not true is the main rotor yoke using the ASHM mission. When evaluating the mini-HUMS data, it can also be compared to the complete HUMS package since the complete HUMS package fatigue life is the closest to the real fatigue life of the part.

As can be seen from the table, for the collective lever, main rotor spindle, and main rotor yoke, in general, the mini-HUMS fatigue lives are reasonably close to, or substantially lower than, the complete HUMS fatigue lives for both the ASHM and GCM missions. The drop in fatigue life for the mini-HUMS is contributed to the fact that mini-HUMS records a limited number of parameters while the complete HUMS unit has more capabilities, including the mission condition recognition. The complete HUMS unit therefore, in general, generates more accurate and reliable data than the mini-HUMS. For the UMMC mission, the mini-HUMS gives lives that are higher than the lives using the complete HUMS package for the rephase lever, the collective lever and the spindle; while for the yoke, the mini-HUMS lives are lower than for the complete HUMS. For the rephase lever with the ASH mission, the mini-HUMS gives fatigue lives that are substantially higher compared to the complete HUMS package. This is also true for the GCM when comparing mini-HUMS concept three to the complete HUMS package.

Thus, it appears that a mini-HUMS can work as an option to a complete HUMS package. However, a mini-HUMS can, in some instances, generate fatigue lives that are actually higher than those generated by the more accurate and complete HUMS package. To account for this, it may be prudent to assign a life extension limit to the part with a mini-HUMS. For example, with a mini-HUMS, the fatigue life of the part could not be extended beyond 200% of the recommended fatigue life using the certification spectrum and as published in the manufacturer's fatigue life report. So, a part that has a recommended life of 5000 hours per the published fatigue life report could be extended up to a maximum of 10,000 hours using a mini-HUMS. This could also be applied to a complete HUMS package, and possibly a sliding scale could be applied, depending on the level of accuracy of the HUMS unit. For example, with only a recording altimeter (concept one), the life could be extended up to a maximum of 200% of the published fatigue life, whereas for a more accurate and complete HUMS package, the life could be extended to a maximum of 400% of the published fatigue life.

5.3 USING GPS IN A HUMS UNIT.

It is anticipated that the GPS will provide data that would allow refinement or replacement of data collected by multiple sensors and would further enhance the accuracy of a mini-HUMS or complete HUMS unit. Typically, a GPS system would provide the aircraft ground track, ground speed, and vertical velocity. These parameters are derived by taking the derivatives of the x, y, and z location of the helicopter in three-dimensional space. These parameters could be used to recognize low-speed and hovering maneuvers as well as improve the turn, climb, and possibly the velocity portions of the HUMS algorithms. GPS data were not recorded during the UMMC, the ASHM, or the GCM.

The current HUMS methodology uses calibrated airspeed to determine aircraft velocity. During low-speed flight (< 50 kts), calibrated airspeed becomes unreliable because of rotor downwash and the low volume of air entering the pitot tube. With the GPS system, using ground track, ground speed, and aircraft heading, it can be determined if the aircraft is in, for example, forward flight, sideward flight, or rearward flight. One drawback is that GPS does not account for wind speed, so the maneuvers recognized are relative to the ground and not the air mass. Because of this, for high-speed flight, it is probably advantageous to use calibrated airspeed and heading. But, for low-speed flight, GPS ground speed and ground track could be used.

A low airspeed circle showing how GPS could be used for low-speed flight recognition is shown in figure 5-2. GPS ground speed, ground track, and vertical velocity along with aircraft true heading, heading rate of change and roll, are used to determine the conditions shown in figure 5-2. Obviously, the resolution and accuracy of a GPS system to be able to recognize the conditions shown in figure 5-2 needs to be high, in the order of resolution and accuracy used in military GPS systems. Possible improvements in existing commercial GPS systems could provide the resolution and accuracy needed to make GPS a viable method for maneuver recognition.

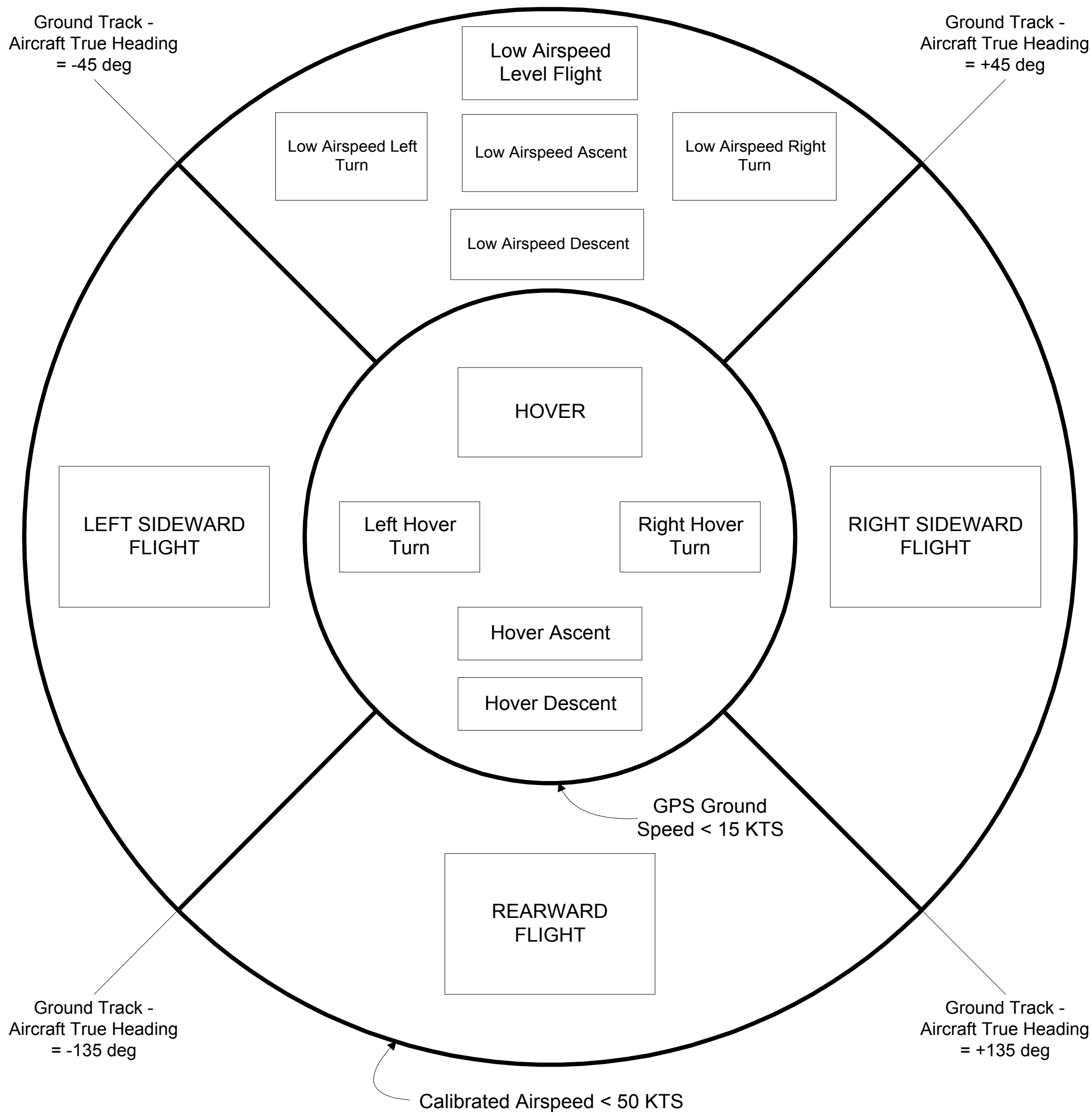

FIGURE 5-2. PROPOSED LOW-SPEED FLIGHT RECOGNITION CIRCLE USING GPS

6. GUIDELINES FOR CERTIFICATION.

6.1 BACKGROUND.

Health and usage monitoring systems are being developed to provide a number of benefits to the helicopter operator. The potential benefits include reduced operating costs, increased availability, and enhanced safety. Some of these potential benefits require FAA credit approval. In these instances, a credit is sought to allow the HUMS to intervene or provide an alternate means of maintaining the continued airworthiness of the aircraft. The certification of HUMS is unique in that the continued airworthiness of the helicopter is improved, and credits are received based primarily on results processed by ground-based equipment.

Ground-based equipment has historically been approved as part of a manufacturer's recommended maintenance procedures. Airworthiness limitations have been approved as part of an aircraft's maintenance manual. The installation of airborne equipment has always been approved as part of the aircraft type certification or supplemental type certification by the appropriate certification office. Since factors that affect the integrity of HUMS processing are now often distributed and include ground-based Commercial-Off-The-Shelf (COTS) equipment and software, both the airborne and ground-based equipment must be certified. Because of this, the selection of a specific system architecture and the determination of hardware and software qualification levels are very important. At the time of this report, no HUMS have been certified for usage credits by the FAA based on the current HUMS Advisory Circular [3].

6.2 HEALTH AND USAGE MONITORING SYSTEM ARCHITECTURE.

Figure 6-1 depicts the end-to-end aspects of a typical HUMS architecture and illustrates the distributed nature of the HUMS equipment and its relationship to existing maintenance equipment. An important consideration is the functional partitioning between airborne and ground-based processing. An additional consideration involves the extent to which the HUMS ground equipment should be integrated with the operator's maintenance management system (MMS). Operators often use PCs and other computer equipment to organize and modernize their maintenance operations. As part of an operator's continuing airworthiness program, the MMS is often used to keep track of aircraft configurations and to derive schedules for maintenance actions, including the removal and replacement of aircraft components or assemblies.

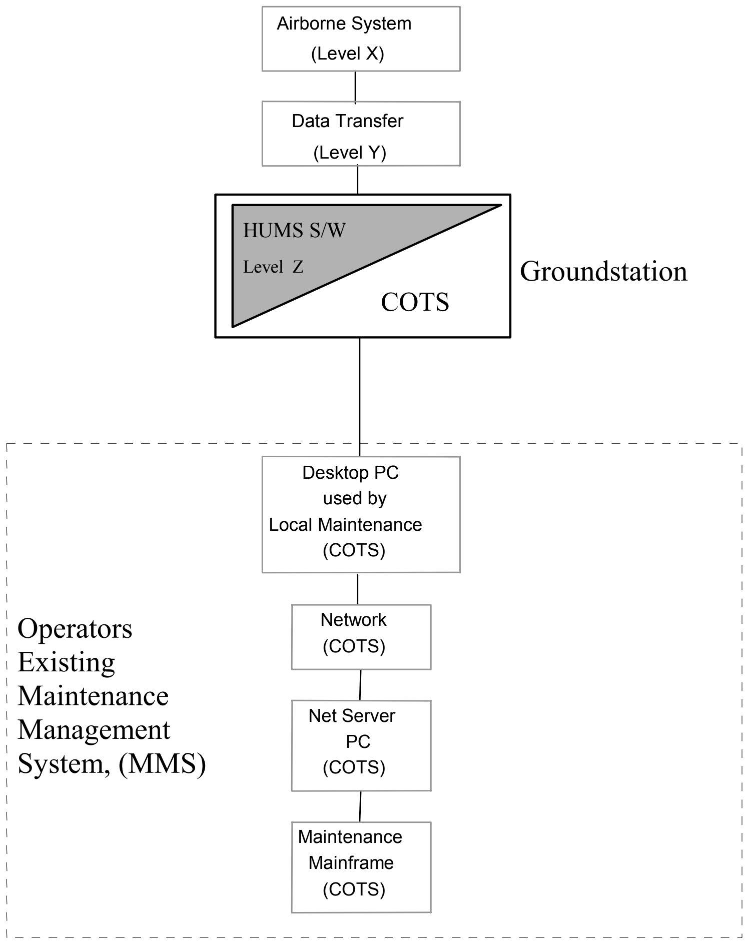

FIGURE 6-1. HEALTH AND USAGE MONITORING SYSTEM ARCHITECTURE

6.3 THE CERTIFICATION PROCESS.

The certification process for HUMS is more complex than traditional certifications because ground-based equipment is usually involved and new technologies are employed. The HUMS certification process has three aspects that are all equally important. These three aspects are installation, credit validation, and continuing airworthiness.

- Installation for a HUMS encompasses all areas of certification required to develop a new system and to install it at an operator's facility. If the system includes a ground-based portion, then that is also included. Everything from airborne equipment design and installation to ground-processing methods and equipment is covered under this aspect of certification.

- Credit validation requires supplying objective evidence that the physics involved in detection, recognition, isolation, or other technology related to the maintenance credit being sought is sufficiently understood.

- Continuing airworthiness documents and demonstrates the operator's ability to successfully operate the HUMS, the operator's procedures and training, the minimum equipment list, how unavailability of the minimum equipment affects the HUMS, and pilot and maintainer actions and procedures.

For each aspect, certain steps are needed to accomplish the certification. Some of these steps will be recognized from traditional certification programs and some are uniquely related to HUMS. The recommended steps for certification based on the HUMS Advisory Circular [3] are listed below.

1. Establish a certification project with the responsible aviation authority
2. Develop an end-to-end system design concept by:

    a. defining the desired maintenance credit(s),

    b. determining the functional partitioning between airborne and ground,

    c. establishing the functional partitioning between HUMS and the maintenance system,

    d. selecting COTS software and hardware with an established service history,

    e. clearly identifying the end of the credit function (algorithm), and

    f. defining a user interface that will meet desired objectives.

3. Prepare and submit hazard assessments for:

    a. airborne installation and
    b. maintenance credits expected or desired.

4. Perform system development in order to:

    a. obtain hardware to meet the system qualification requirements and
    b. establish application software to the required DO-178B levels.

5. Test the application in the COTS environment.
6. Validate the COTS using an independent means of verification.
7. Develop a user operating manual for the system defining credit requirements.
8. Modify maintenance and/or flight manuals for the proposed credits.
9. Certify the airborne installation.
10. Conduct a controlled service introduction for credit validation.
11. Helicopter operator to obtain credit approval for his aircraft.

The certification of a HUMS for usage credits will require that a number of processing and integrity checks be included in the overall HUMS to ensure the validity of the data and calculations at all times. There are two factors that can directly affect the correctness of the flight spectrum and, therefore, the accumulated damage rates. These include the possibility of missing data or inaccurate data. In an operational system, two reports, the process report and release report, are intended to deal with these two possibilities.

The following list contains an outline of a proposed end-to-end process for usage monitoring to deal with the possibility that these operational problems will occur. The outline includes preflight, postflight, ground-based processing, and part replacement. These steps are a guide to the elements that should be included in the operator's maintenance manual procedures for usage credits.

1. Install a PC card in the HUMS-equipped aircraft.
2. Conduct scheduled flight operations.
3. Remove PC card, deliver to usage ground station.
4. Perform PC card download operation.

    a. Transfer PC card data to ground station.
    b. Confirm data copy; prepare the PC card for next use.

5. Process data from memory card.

    a. Check for unreasonable, invalid, or missing data; apply engineering units.
    b. Run flight condition recognition algorithms.

        i. For each maneuver identified, run an appropriate parameter correlation module.

        ii. Create an interval spectrum for each operation.

    c. Save the operations interval spectrums until the next data release.
    d. Produce a process report for the PC card data that were downloaded.

        i. List any anomalies and actions that should be taken.

        ii. From parameter correlation function, list any flight data parameters that should be audited.

        iii. Check for excessive time in the "unrecognized" flight regime[1].

        iv. Include message reminding of any audits, checks, or system calibrations that are due.

6. If a Flight Parameter Calibration Audit was performed, input the results.
7. If a system integrity check was performed, have user input the results.
8. When an aircraft logbook audit is performed, make sure all flight hours are accounted for.

[1] Ask for full system parameter audit if unrecognized data is beyond limits.

9. Release data to maintenance data-tracking facility.

   a. Gap fill for missing data (including where data was found invalid or unreasonable).

   b. Compute life expended for all operations since the last release of data.

   c. Trend the damage rate and take appropriate action[2].

   d. Produce release report. List each operation processed. Include:

      i. Operation time.
      ii. Number of flights.
      iii. Damage to parts for each operation.

10. Activate maintenance data-tracking facility.

   a. Receives interval component damage data.
   b. Accumulates component damage over time.
   c. Reports to maintainer when part should be replaced.

These procedures should be followed during the certification process and also reviewed during a controlled service introduction to validate the suggested process. It is recommended that a demonstration program be conducted to evaluate a usage credit system. The suggested demonstration program would require the involvement of helicopter manufacturers, the FAA, and a commercial operator. Such a program could validate the application of the HUMS Advisory Circular. The application of usage monitoring has clear benefits to achieve reduced operating costs and enhanced safety through the monitoring of individual aircraft usage.

[2] If unacceptable shift in trend, require pilot(s) and/or maintainer(s) concurrence to confirm mission severity change. If there are no mission or aircraft changes, then require full parameter audit.

7. DAMAGE TOLERANCE INVESTIGATION.

7.1 THEORETICAL REDESIGN OF FOUR PSEs TO MEET DAMAGE TOLERANCE REQUIREMENTS.

This section discusses the unique characteristics of high-cycle fatigue in helicopters and reviews the results of the theoretical redesign of the four PSEs to meet damage tolerance requirements. This study compares the original safe-life design to the theoretical damage tolerance design, including differences in component weight. The basic fatigue methodologies of safe life and damage tolerance are also discussed.

Under a previous study [1], a damage tolerance analysis was performed on the four selected PSEs. The results of the previous study are presented in tables 7-1 and 7-2 for the certification mission, the GCM, and the ASHM. The crack growth lives were developed using CRKGRO, a software package designed specifically to perform this task [4 and 5]. As can be seen, for a 0.015-inch initial flaw, the flight hours to critical crack length are very low for all three missions. The UMMC spectrum, in terms of severity, is between the GCM and the ASHM. Thus, it was assumed that the flight hours to critical crack length for the UMMC mission would be comparable to those shown below. A theoretical redesign of the four PSEs was conducted to provide acceptable inspection intervals using damage tolerance methodologies. The results of the theoretical redesign are discussed in detail in this section.

TABLE 7-1. FLIGHT HOURS TO CRITICAL CRACK LENGTH—0.005-inch INITIAL CRACK

	Certification Mission	Gulf Coast Mission	Atlanta Short Haul Mission
Rephase Lever	No Growth	No Growth	No Growth
Collective Lever	192	271	554
Main Rotor Spindle	No Growth	No Growth	No Growth
Main Rotor Yoke	160	7790	2910

TABLE 7-2. FLIGHT HOURS TO CRITICAL CRACK LENGTH—0.015-inch INITIAL CRACK

	Certification Mission	Gulf Coast Mission	Atlanta Short Haul Mission
Rephase Lever	78	259	154
Collective Lever	13	16	31
Main Rotor Spindle	143	104	2557
Main Rotor Yoke	20	50	70

Most current helicopters are certified using the safe-life fatigue methodology. The damage tolerance methodology may also be implemented for use in helicopter certification when a crack is assumed to exist in the part and the growth of the crack is predicted. Note that usage monitoring can be used in conjunction with both safe-life and damage tolerance evaluations of helicopter components.

Summarized below are the results of a study that developed theoretical redesigns of four helicopter components in order to configure these components to potentially meet damage tolerance requirements. In the following discussion, it is important to recognize that each of the four PSEs had to maintain the fit and function of their baseline part. Also, while the aluminum and titanium baseline parts had shot-peened surface treatments, no credit was given for shot peening in the damage tolerance analysis. It is also important to note that the theoretical results shown are based on hypothetical assumptions for the purposes of this study (i.e., to directly compare safe-life methodology to damage tolerance) and do not necessarily reflect the actual loading of the baseline parts on the helicopter. Consequently, the fatigue lives and inspection intervals determined for purposes of this study should not be used to draw any conclusions concerning certification or airworthiness of any helicopter.

7.2 FATIGUE METHODOLOGY.

Helicopter dynamic components (rotors and controls) operate in a high-cycle fatigue environment where every rotation of the helicopter blade in flight causes one or more fatigue cycles on flight-critical rotating components. Helicopters also have ground-air-ground (GAG) low-cycle fatigue loading, where one GAG load cycle is defined by an excursion to the highest load seen in flight and back to a load where the helicopter is stationary on the ground. Two approaches to fatigue certification are the safe-life and damage tolerance methods that are summarized below.

7.2.1 Safe-Life Method.

For most current helicopters, rotating dynamic components (rotors and controls) are certified for fatigue using the safe-life methodology. Based on extensive service history, the safe-life methodology has proven to be a satisfactory approach for fatigue design. In this approach, the oscillatory fatigue stresses measured in flight are compared to an oscillatory endurance limit established for the part using component oscillatory stress versus cycles to failure (S-N) fatigue test data. An example of an S-N curve is shown in figure 7-1.

All measured flight stresses above the reduced endurance limit produce some fatigue damage. Using Miner's rule, a fatigue life, in flight hours, is calculated using the flight stresses and the certification flight spectrum developed for the aircraft. A fatigue life is established by assessing the frequency and magnitudes of oscillatory stresses above the reduced endurance limit. A retirement life can then be established for the part, so that the part can be removed from service before the safe life of the part has been reached.

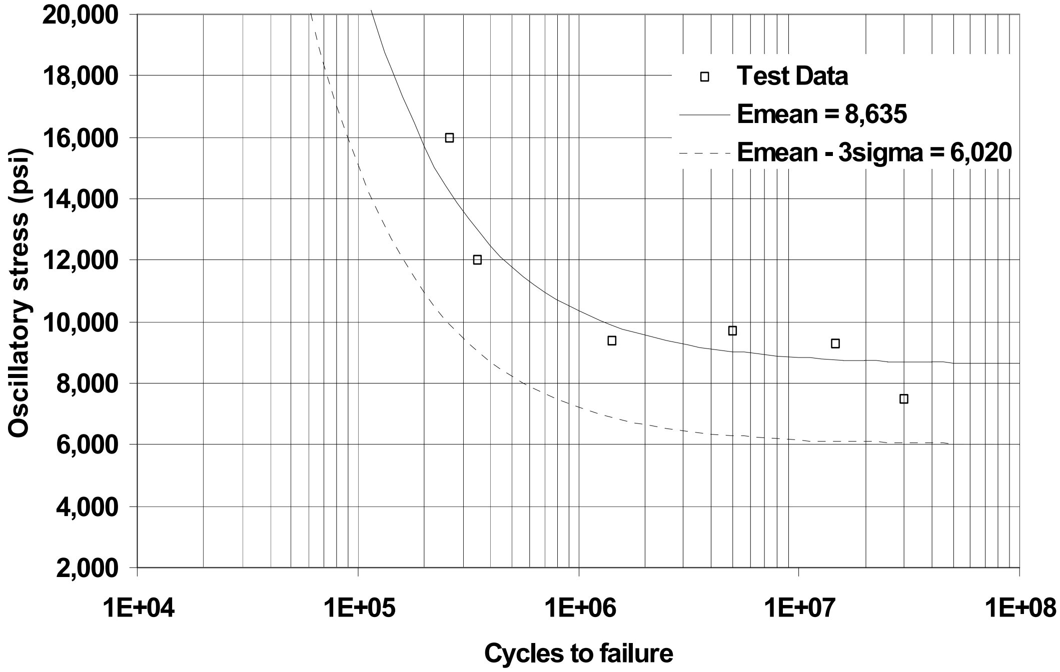

FIGURE 7-1. TYPICAL S-N CURVE

<u>7.2.2 Damage Tolerance Method</u>.

The damage tolerance approach assumes a crack or flaw to exist in the part. Typically, in rotorcraft applications, the crack length is established as a 0.015-inch semicircular flaw. This approach offers the potential for enhanced safety in the unlikely event that a crack occurs in a critical area. Using crack growth data for the component and geometry, the flight stresses for both high- and low-cycle fatigue loading are compared with a stress that would cause the crack to grow. This comparison is made by calculating the stress-intensity factor for a given load condition and comparing it with the threshold stress-intensity factor for the material. This methodology is typical of one that would be used for damage tolerance certification of helicopter components. For a situation where the stresses will cause the crack to grow, the time interval for the original flaw to grow to the critical crack length (failure of the part) is calculated using a flight spectrum developed for the aircraft. This time interval is called the crack growth life. An inspection interval is then established by subdividing the crack growth life so that a detailed inspection will be able to find cracks in the part before they grow to the critical length.

For the purposes of this study, it was assumed that the inspection interval is equal to one-half of the calculated crack growth life of the part. Figure 7-2 shows a typical damage tolerance analysis procedure used to design a part for damage tolerance. In this figure, ΔK is defined as the stress-intensity factor range and ΔK_{TH} is defined as the threshold stress-intensity factor range. Per standard fracture mechanics methodology, a ΔK above ΔK_{TH} will cause a crack to grow.

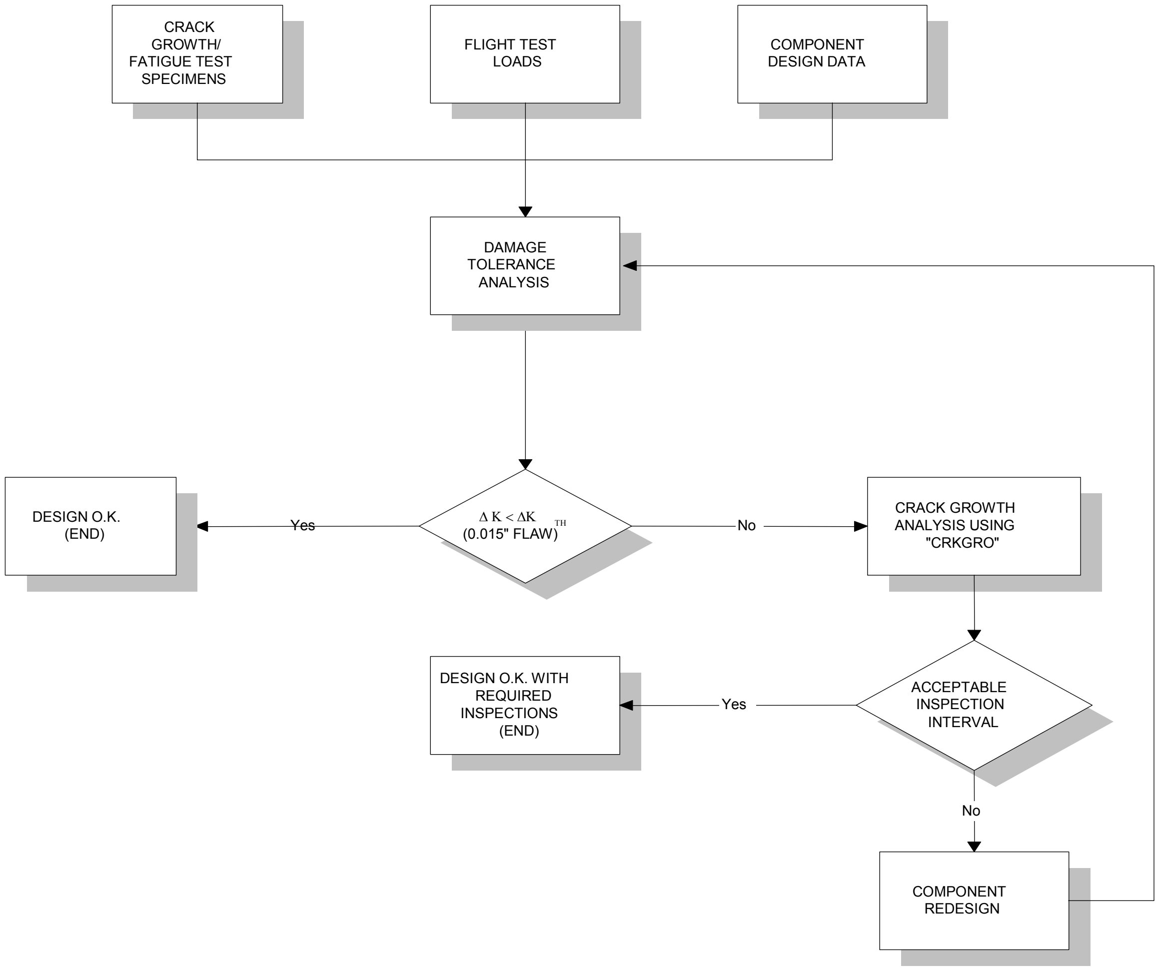

FIGURE 7-2. DAMAGE TOLERANCE ANALYSIS PROCEDURE

Figure 7-3 shows a typical *da/dN* curve, where *da/dN* is defined as the amount of crack growth per cycle. This curve is used to establish the threshold stress-intensity factor for the material as well as to calculate a crack growth life. The crack growth data used in this study primarily came from published crack growth data, such as that shown in references 6 through 8. Note that this kind of curve is valid only for the specific material, orientation, temperature, environment, loading frequency, and possibly other pertinent factors involved in the testing that produced it.

Material:	**7075 Aluminum**
Heat Treatment:	**T7351**
Product Form:	**Plate & Sheet**
Orientation:	**T-L**
Environment:	**Room Temperature LA & DA**

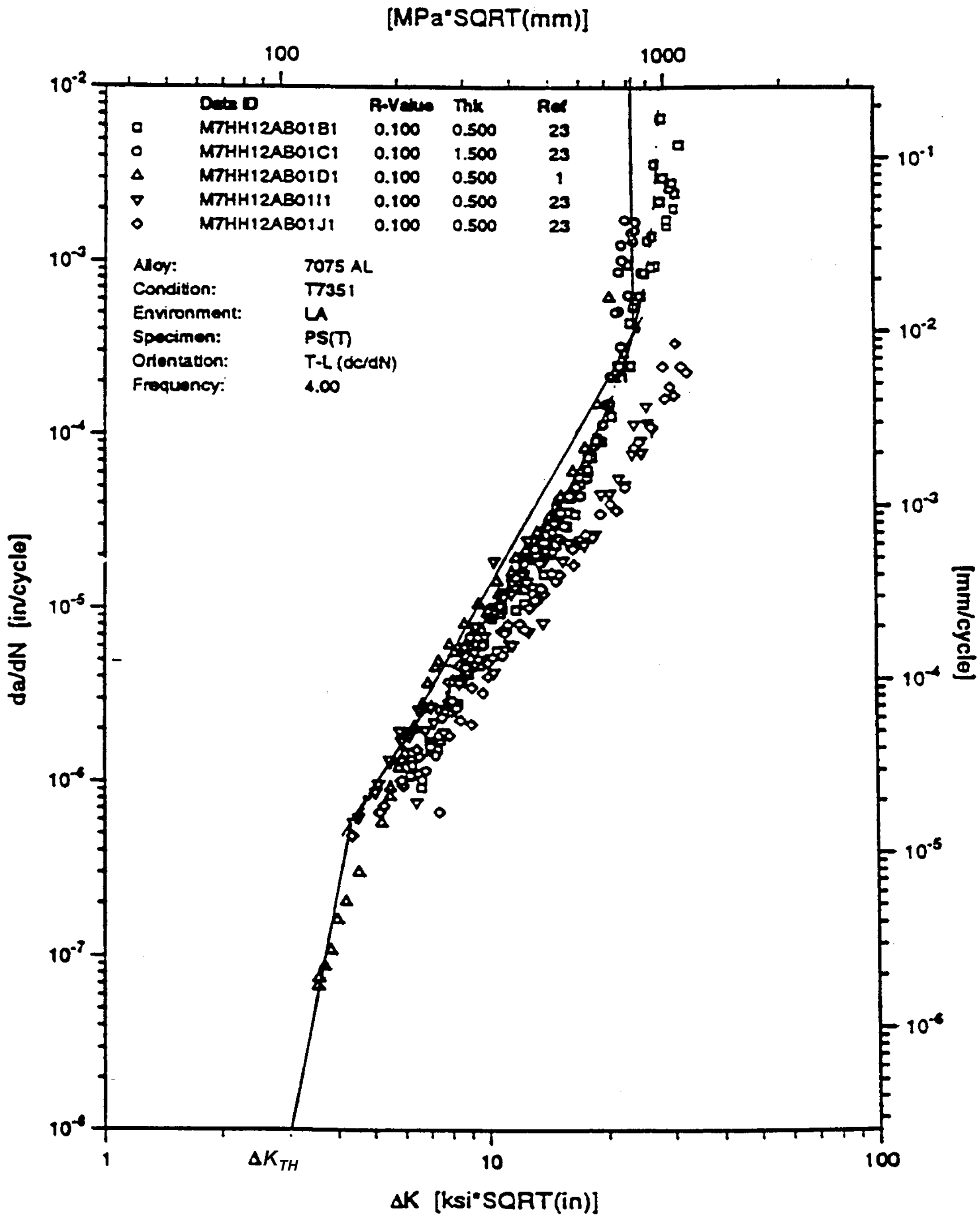

FIGURE 7-3. TYPICAL *da/dN* DATA FOR 7075-T7351 ALUMINUM ALLOY IN *T–L* DIRECTION

7.3 DAMAGE TOLERANCE INVESTIGATION OF FOUR PSEs.

The four baseline PSEs addressed in this study were selected for theoretical redesigns to potentially meet damage-tolerant requirements. (Figure 3-1 shows these components as part of the hub and blade assembly of the 412 helicopter.) These components were chosen to provide a variety of materials and applications data. The materials used in the selected components are as follows:

- Main rotor yoke — titanium (6Al-4V)
- Main rotor spindle — steel (15-5 PH)
- Main rotor rephase lever — aluminum (7075-T73)
- Collective lever — aluminum (7075-T73)

These components were designed and certified based on a safe-life approach and were assigned a specific retirement life. Theoretical redesigns of the four PSEs were made in critical areas to investigate the potential weight impact of meeting damage tolerance requirements. The goal was to attain a no-growth status for each PSE for the most critical flight load from the certification load survey, thus incurring no additional inspections and no increased maintenance cost.

The damage tolerance and weight impact calculations were based on the certification spectrum for all four PSEs. The critical areas evaluated for each PSE are summarized below.

- The main rotor yoke was evaluated at station 4.8 because this was the predominant failure location in the fatigue testing.

- The main rotor spindle was evaluated in a highly stressed area of the spindle lugs where the main rotor blade is attached.

- The main rotor rephase lever was modified in areas where the calculated stress was above the threshold value.

- The main rotor collective lever was modified in areas where the stress was above the threshold value.

These parts were theoretically redesigned for no crack growth, if possible, or for a crack growth life at least twice the retirement life of the part, thus essentially eliminating the need for a specific damage tolerance inspection. The theoretical changes included geometry changes or material changes, as discussed for each of the four selected components in the following sections.

7.3.1 Main Rotor Yoke.

The baseline main rotor yoke geometry is shown in figure 7-4. The main rotor yoke is made from 6Al-4V titanium plate using the BSHTOA process, which significantly improves the threshold stress-intensity factor range (ΔK_{TH}) for the material to a ΔK_{TH} = 5.5 ksi (in)$^{1/2}$. This can be compared to a standard annealed plate for which ΔK_{TH} = 3.75 ksi (in)$^{1/2}$. If the maximum stress experienced by the part produces a ΔK below ΔK_{TH}, then a given crack should not grow according to fracture mechanics. Thus, it is advantageous to have a high ΔK_{TH}. In the following analysis, a value of ΔK_{TH} = 5.5 ksi $(in)^{1/2}$ was used.

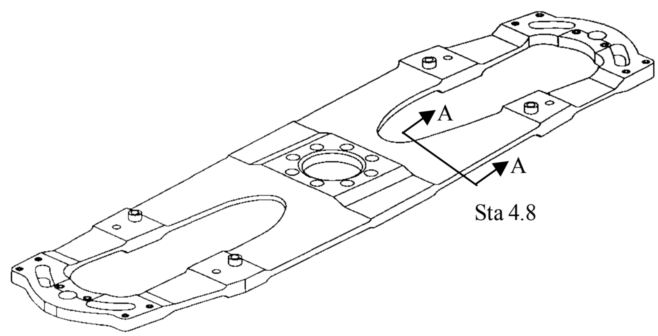

FIGURE 7-4. MAIN ROTOR YOKE GEOMETRY

The baseline main rotor yoke has a safe-life retirement time of 5000 hours and a theoretical damage tolerant crack growth life of 20 hours for a 0.015-inch initial flaw size (see table 7-2). For the theoretical redesign, the main rotor yoke was analyzed in the flexure area at span station 4.8, since this was the predominant failure location from fatigue testing and was used for damage tolerance analysis in reference 2. Figure 7-5 shows a representative cross section of the yoke flexure that was used for analysis, assuming a corner crack in the yoke flexure. This analysis assumes that each side of the flexure has the cross section, shown below, and that each shares the applied load equally.

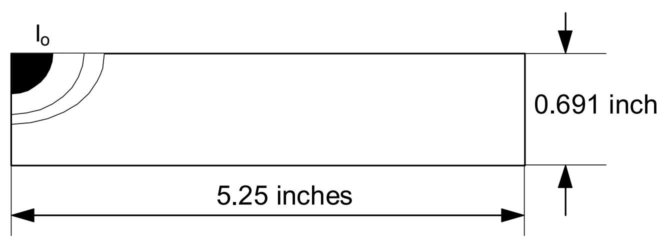

FIGURE 7-5. MAIN ROTOR YOKE SECTION A-A AT STATION 4.8

Note that the flexure thickness varies down the length of the yoke. This analysis assumes that down the length of the yoke, the flexure thickness is increased by the same percentage so that the critical fatigue location remains at station 4.8.

An attempt was made to increase the flexure thickness so that the maximum stress in the flexure was below the stress corresponding to a ΔK_{TH} of 5.5 ksi $(in)^{1/2}$. Assuming a steady stress approximately equal to the oscillatory stress, the peak stress corresponding to ΔK_{TH} = 5.5 ksi $(in)^{1/2}$ is 36,265 psi. The yoke flexure thickness was theoretically increased in an attempt to reduce the peak stress in the flexure to below 36,265 psi for the highest load maneuver from the certification load-level survey. A plot of this is shown in figure 7-6.

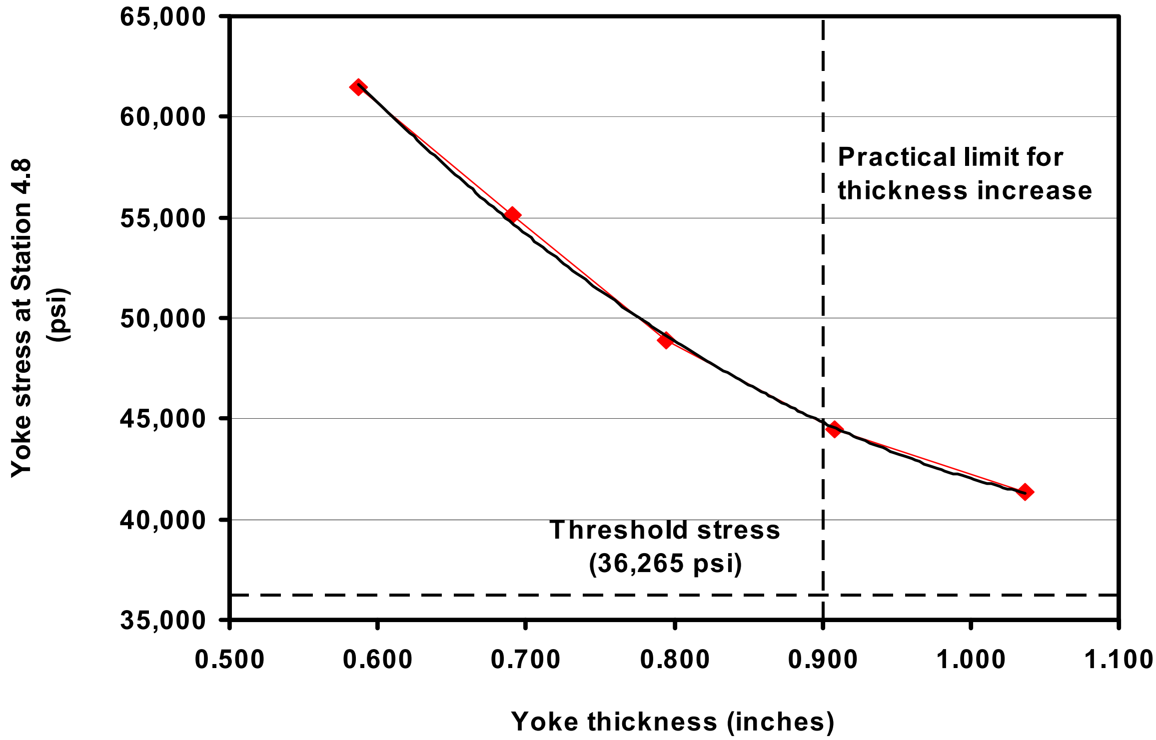

FIGURE 7-6. YOKE STRESS VERSUS THICKNESS

At the original thickness of 0.691 inch, the peak stress was 55,111 psi. This is well above the stress that corresponds to the threshold, which is 36,265 psi. With a 31.4% increase in thickness to 0.908 inch, the peak stress becomes 44,470 psi for the most severe maneuver. As can be seen in figure 7-6, as the thickness increases, the stress decreases at a slower rate. This is because of yoke stiffness; i.e., as the thickness is increased, the flexure becomes stiffer and carries more load for the same flapping displacement of the main rotor. Thus, the stress decreases at a slower rate than would be anticipated, assuming a constant load application. To increase the yoke thickness so that the stress is below the threshold stress of 36,265 psi becomes impractical, if not infeasible. A flexure thickness increase of approximately 30% (0.90 inch thickness) is considered a practical limit for the yoke flexure, considering stiffness increases, possible dynamic response effects, and weight.

Based on the theoretical results showing an unacceptable stiffness increase as the thickness increases, it does not appear that the main rotor yoke can be practically redesigned so that the crack growth life is twice the retirement life of the part. An alternative to the approach of increasing the thickness of the titanium yoke might be to change the material to a composite material. Because of the superior damage tolerance capabilities of composite materials, a composite main rotor yoke might be an alternative to a titanium yoke to meet damage tolerance certification requirements. Obviously, evaluation of such an alternative would require additional research and testing.

7.3.2 Main Rotor Spindle.

The baseline main rotor spindle geometry is shown in figure 7-7. This component has a safe-life retirement time of 10,000 hours and a theoretical damage-tolerant crack growth life of 143 hours for a 0.015-inch initial flaw size (see table 7-2). For the theoretical redesign, the main rotor spindle was analyzed for damage tolerance in the spindle lugs where the main rotor blade is attached. The goal was to theoretically redesign the spindle lugs so that they would incur no crack growth using a 0.015-inch initial flaw size for the most severe maneuver in the certification load-level survey. A cross section through the spindle baseline lugs is shown in figure 7-8.

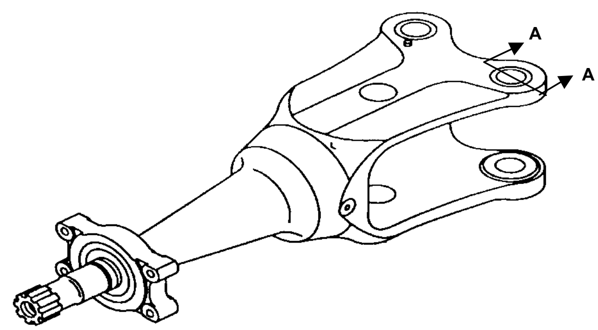

FIGURE 7-7. MAIN ROTOR SPINDLE

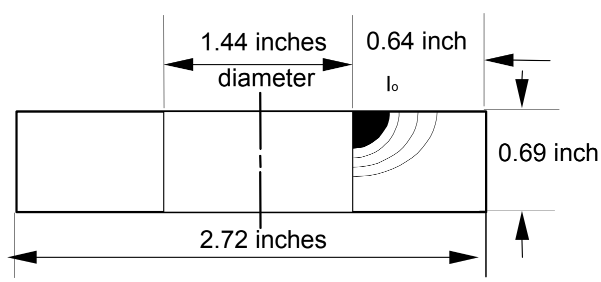

FIGURE 7-8. MAIN ROTOR SPINDLE SECTION A-A

In this analysis it was assumed that the spindle was made from 15-5 stainless steel, which has a significant improvement in ΔK_{TH} compared to 4340 stainless steel. For 15-5 stainless steel, $\Delta K_{TH} = 5.0$ ksi (in)$^{1/2}$ compared to 3.12 ksi (in)$^{1/2}$ for 4340 steel. The cross-sectional area through the spindle lugs was increased until all flight maneuvers produced a ΔK below ΔK_{TH}.

The maximum and minimum spindle lug loads from the certification load-level survey were 18,871 lb and 9,623 lb, respectively. The GAG load is a peak of 18,871 lb to zero and back, or 9435.5 ±9435.5 lb.

The spindle lugs were evaluated using several scenarios to identify the option that provided the least amount of weight increase. The candidate options were increasing the thickness and width of the lugs by the same percentage, increasing the thickness by a larger percentage than the width, and increasing only the thickness.

Using the worst-case flight maneuver lug load excursion from a maximum peak of 18,871 lb to a minimum of 9,623 lb, the lug was theoretically redesigned so no growth occurs, assuming a 0.015-inch flaw. It was determined that increasing the thickness of the lug provided the greatest gain in damage-tolerant capabilities for a given weight increase. Also note that this thickness increase only applies to two of the four lugs of the spindle (the upper aft lug and the lower forward lug), because at these lug locations, the chord and beam moments are in phase. At the lower aft and upper forward lugs, the out-of-phase moments subtract from one another, making these lugs significantly less critical in fatigue and fracture. The theoretical redesign cross section is shown in figure 7-9 for the upper aft and lower forward lugs.

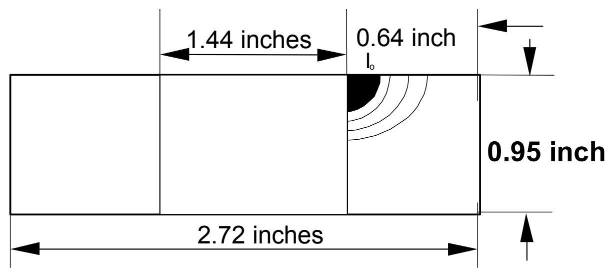

FIGURE 7-9. MAIN ROTOR SPINDLE SECTION A-A AFTER THEORETICAL REDESIGN

Note that the lug thickness was increased from 0.69 to 0.95 inch, a 38% increase. This increase was assumed to be away from the blade interface so that the blade would require no change. However, the blade-to-spindle bolts would have to be made longer for this change. Also note that the GAG loading of 9435.5 ±9435.5 lb would cause an 0.015-inch flaw to grow. But because GAG loading is infrequent (assumed to be four GAG cycles per hour for this analysis), the crack growth life on the spindle is still greater than twice the retirement life of the part, or 20,000 hours.

To make a 0.015-inch flaw in the spindle lug fall below the crack growth threshold for GAG loading, a 1.14-inch-thick lug is required. This is a rather dramatic increase over the 0.95 inch thickness that is required for flight loading. For a 0.95-inch-thick lug, the approximate calculated weight increase per helicopter is 4.5 lb, assuming this change applies to two lugs per spindle, assuming longer bolts, and assuming some added material around the thicker lugs to taper the additional thickness back in the basic section of the spindle. Note that this weight increase of 4.5 lb is per helicopter, or 1.13 lb per spindle. Each spindle assembly currently weighs 24.5 lb, so the 1.13 lb represents approximately a 4.6% weight increase.

Note that since the spindle was theoretically redesigned to the certification spectrum loads and assuming that none of the HUMS missions would require that the helicopter perform more than four GAG cycles per hour over the life of the aircraft, the theoretical redesign shown would apply to all missions described in this report. For a helicopter operating with more than four GAG cycles per hour over the life of the aircraft, the theoretical redesign would have to be reanalyzed since the inspection interval of 10,000 hours was based on four GAG cycles per hour.

7.3.3 Collective Lever.

The baseline collective lever geometry is shown in figure 7-10. The baseline collective lever has a safe-life retirement time of 10,000 hours and a theoretical damage-tolerant crack growth life of 13 hours for a 0.015-inch initial flaw size (see table 7-2). For the theoretical redesign, the collective lever was analyzed using an ANSYS finite element model to determine those locations in the collective lever that produced a stress above the threshold stress for crack growth. Additional material was added in these areas to reduce the stress below the threshold stress for crack growth for the highest load condition in the certification spectrum. For 7075-T73 aluminum, ΔK_{TH} is equal to 2.5 ksi $(in)^{1/2}$. Assuming a steady stress approximately equal to the oscillatory stress, the peak stress corresponding to ΔK_{TH} = 2.5 ksi $(in)^{1/2}$ is 13,000 psi for a 0.015-inch initial flaw size, or 6500 psi ±6500 psi. This stress is based on an assumed corner crack in a plate, which is the most representative crack geometry for the collective lever.

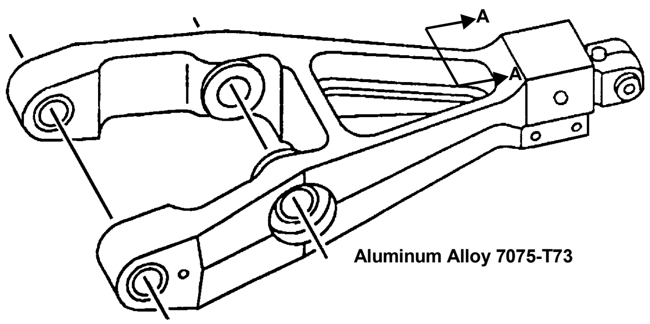

FIGURE 7-10. COLLECTIVE LEVER

In figure 7-11, section layouts are shown for the baseline collective lever design. Figure 7-12 shows section layouts for the theoretically redesigned collective lever. The highest stress in the collective lever is seen at section A-A in figure 7-10, near where the collective boost tube connects to the collective lever.

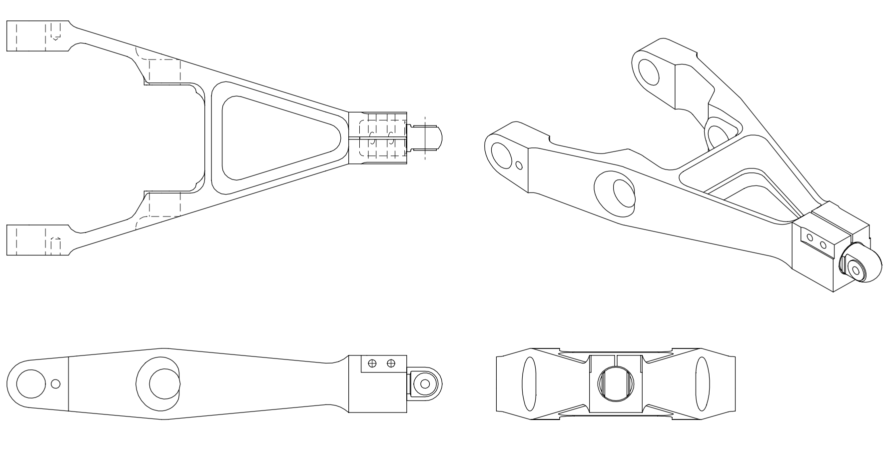

FIGURE 7-11. BASELINE DESIGN OF COLLECTIVE LEVER

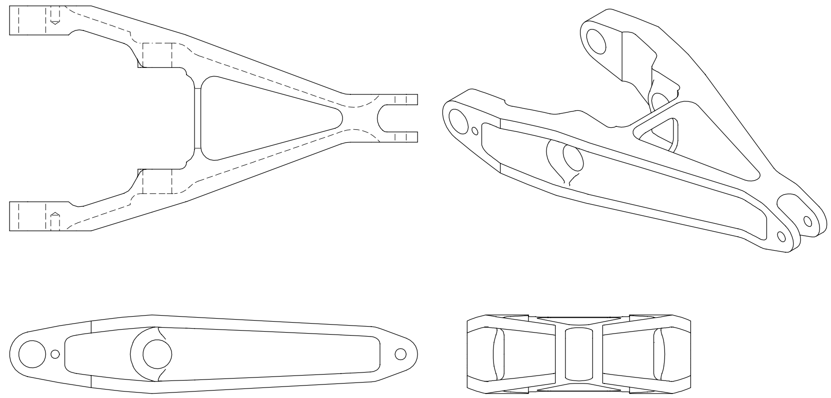

FIGURE 7-12. THEORETICAL REDESIGN OF COLLECTIVE LEVER FOR DAMAGE TOLERANCE

The baseline collective lever assembly weighs a total of 4.36 lb, which includes the detail part (3.26 lb); peripheral hardware such as bushings, washers, and bolts (0.41 lb); and the rod end assembly, which is bolted to the top A of the collective lever and attaches to a clevis on the collective boost tube (0.69 lb). The projected redesign (to theoretically meet damage tolerance requirements) weighs 5.32 lb and includes the detail part (4.91 lb) and the peripheral hardware (0.41 lb), which equates to a 22% increase in weight.

Notice that in the analytical redesign, the rod end assembly has been eliminated and an integral clevis has been put in its place. This design is cleaner and more efficient from a weight standpoint, but it would require the current clevis on the end of the collective boost tube be replaced with a rod end similar to the one in the current collective lever. The weight change to the collective boost tube is assumed to be minimal for this change, since a rod end should weigh approximately the same as a corresponding clevis. Note that since the collective lever was theoretically redesigned to the certification spectrum loads, the theoretical redesign shown would apply to all missions described in this report.

7.3.4 Rephase Lever.

The baseline rephase lever geometry is shown in figure 7-13. This component has a safe-life retirement time of 5000 hours and a theoretical damage-tolerant crack growth life of 78 hours, assuming a 0.015-inch initial flaw (see table 7-2). For the theoretical redesign, the rephase lever was analyzed using an ANSYS finite element model to determine those locations in the rephase lever that produced a stress above the threshold stress for crack growth. Additional material was added in these areas to reduce the stress below the threshold stress for crack growth for the highest load condition in the certification spectrum. For 7075-T73 aluminum, ΔK_{TH} = 2.5 ksi $(in)^{1/2}$. Assuming a steady stress approximately equal to the oscillatory stress, the peak stress corresponding to ΔK_{TH} = 2.5 ksi $(in)^{1/2}$ is 13,000 psi for a 0.015-inch initial flaw size, or 6500 psi ±6500 psi. This stress is based on an assumed corner crack in a plate, which was assumed to be the most representative crack geometry for the rephase lever.

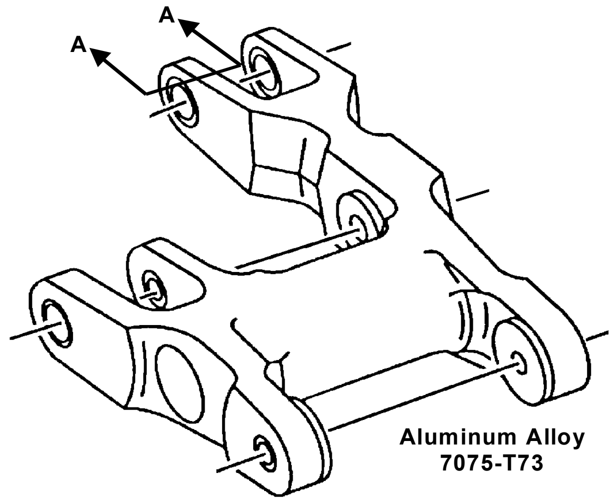

FIGURE 7-13. REPHASE LEVER GEOMETRY

In figure 7-14, section layouts are shown for the baseline rephase lever design. Figure 7-15 shows section layouts for the theoretical redesign of the rephase lever. As can be seen from the layouts, additional material was added to the rephase lever arm that attaches to the main rotor pitch link, as this was the area with the highest stress level. The baseline rephase lever assembly weighs a total of 3.34 lb, and the theoretical redesign (to meet damage tolerance requirements) weighs 3.84 lb, which equates to a 15% increase in weight. The rephase lever assembly includes the weight of the rephase lever detail (3.07 lb) and the weight of several bushings and an insert (0.27 lb). The theoretical redesign detail weighs 3.57 lb for a total weight of 3.84 lb for the rephase lever assembly. Note that since the rephase lever was theoretically redesigned to the certification spectrum loads, the theoretical redesign shown would apply to all missions described in this report.

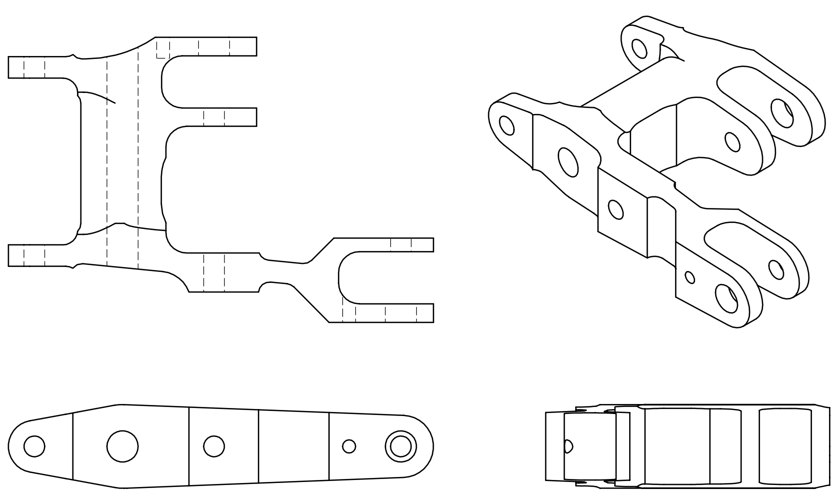

FIGURE 7-14. BASELINE DESIGN OF REPHASE LEVER

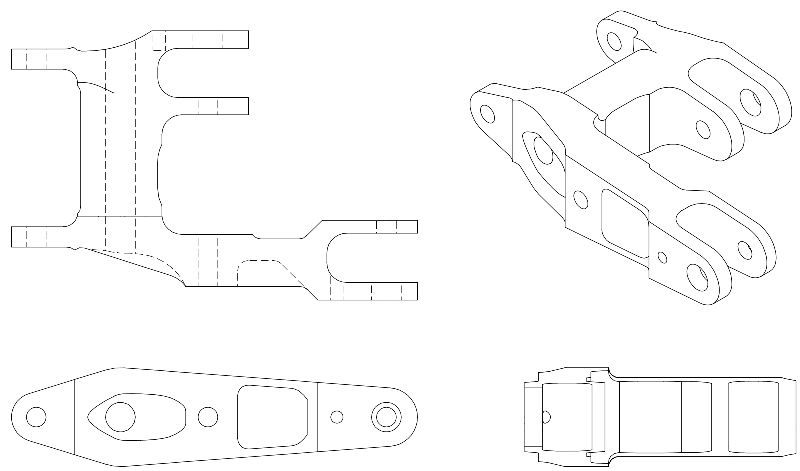

FIGURE 7-15. THEORETICAL REDESIGN OF REPHASE LEVER TO MEET DAMAGE TOLERANCE REQUIREMENTS

7.4 SUMMARY OF DAMAGE TOLERANCE RESULTS.

Helicopter dynamic components, specifically rotors and rotating controls, operate in a high-cycle fatigue environment. Most current certified helicopters use a safe-life (no-flaw) fatigue methodology to establish retirement times for dynamic components. Damage tolerance methodology (initial flaw or crack assumed) is used to define crack growth inspection intervals and is an alternative method for certification of helicopter dynamic components. Four PSEs were selected to evaluate the effect of a theoretical damage tolerance analysis on parts designed for safe life—the main rotor yoke, the main rotor spindle, the collective lever, and the rephase lever.

The baseline parts were made from titanium, steel, and aluminum, and all required a theoretical redesign to get an acceptable damage tolerance inspection interval. The projected weight increases ranged from 4.6% to 22%. Although the weight penalty was significant for the collective lever and the rephase lever, it is expected that, for an optimized new design that incorporates damage tolerance methodology from the beginning of the development process, the weight impact would be less. On parts with high steady loading and significant oscillatory loading, such as seen in the main rotor yoke, composite materials or redundant load path designs are believed to be more effective from a weight standpoint to meet damage tolerance requirements.

Table 7-3 is a summary of the results of the theoretical damage-tolerant redesigns together with comparisons to the retirement life of the four baseline parts using a safe-life fatigue calculation methodology. The table specifically shows (1) the calculated crack growth life of the theoretical redesigns using a 0.015-inch initial flaw size, (2) the corresponding inspection interval, if applicable, for the redesigns, assuming an inspection interval equal to one-half the calculated crack growth life, (3) the material used for each part, and (4) the percentage weight increase calculated for each part to meet the damage-tolerant crack growth life and corresponding inspection interval.

TABLE 7-3. SUMMARY OF DAMAGE TOLERANCE RESULTS FOR FOUR PSEs

Helicopter PSE	Baseline Safe-Life Retirement Time	Baseline Damage-Tolerant Crack Growth Life (see table 7-2)	Calculated Damage-Tolerant Crack Growth life[a]	Calculated Damage-Tolerant Inspection Interval[a]	Material	Weight Increase Over Baseline Assembly (%)
Main rotor yoke	5,000 hrs	20 hrs	N/A[b]	N/A[b]	6Al-4V titanium with BSHTOA	N/A[b]
Main rotor spindle	10,000 hrs	143 hrs	> 20,000 hrs	10,000 hrs	15-5 stainless steel	4.6%
Collective lever	10,000 hrs	13 hrs	No crack growth	No inspection required	7075-T73 aluminum	22%
Rephase lever	5,000 hrs	78 hrs	No crack growth	No inspection required	7075-T73 aluminum	15%

[a] Crack growth life based on limited analytical study results for theoretical redesigns of each PSE.

[b] Composite material might be an alternative for the main rotor yoke to potentially meet damage tolerance requirements with a 5000-hour inspection interval.

Note that the weight increase for the control system or rotor would be less than the weight increase for the individual parts shown, since certain structural elements, such as bearings, bolts, or bushings, will not change as a result of damage tolerance requirements. Note also that the theoretical analysis shown is based on certain assumptions for the purposes of this study to directly compare safe-life methodology to damage tolerance and does not reflect the actual loading of these parts on a production helicopter.

8. SUMMARY AND CONCLUSIONS.

The usage monitoring from the utility mission in Morgan City (UMMC) provided additional data to validate the use of a health and usage monitoring system (HUMS). The UMMC usage data indicated a mission similar to the Gulf Coast mission (GCM) with a significant portion of time spent at cruise speed in level flight. The UMMC is at low altitude (< 3000 ft) for most of the time (76%) and was similar to the GCM (61% < 3000 ft). The percentage of time at heavy gross weight (GW) was high (95%). It is suspected that the GW was not entered routinely since the GW unit defaults to high GW if not entered. This draws attention to the importance of standard operating procedures to maximize the benefit of a HUMS and emphasizes the benefit of an automatic GW measurement system if it can be developed to work accurately, routinely, and without significant additional cost to the HUMS unit. Additionally, the default GW should be set to a value that clearly indicates whether the pilot entered the GW or not.

The flight condition recognition software was able to recognize the maneuvers associated with the UMMC operation. The percentage of unrecognized data was extremely low (0.011%). The recorded cyclic and collective boost tube oscillatory loads during the Atlanta short haul mission (ASHM) did indicate conservatism in the certification load survey loads; however, when the measured ASHM GW, altitude, and center of gravity breakdown is applied to the certification loads, the certification loads get closer to the measured ASHM loads, indicating that it may be prudent to limit how refined the HUMS spectrum is made and how refined the certification load-level survey is flown.

The main conclusions that can be drawn from this study are as follows.

1. To maximize the benefit of a HUMS, it may be advantageous to fly the certification load survey for a new helicopter with more intermediate points and have the HUMS unit programmed to fit maneuvers into these intermediate categories. However, the additional cost to the load surveys needs to be considered as well as what level of refinement is desired from the load survey as mentioned previously.

2. Guidelines for certification and maintenance concepts were developed for HUMS implementation. The next step would be to validate this process through a Controlled Service Introduction demonstration program.

3. A mini-HUMS offers potential benefit as a low-cost alternative to a complete HUMS package. However, with this kind of system, the decreased accuracy needs to be considered when analyzing and evaluating the HUMS data.

4. The integration of a global positioning system (GPS) into a HUMS presents a viable opportunity for improved maneuver recognition; both for a complete HUMS package as well as a mini-HUMS. The use of GPS with HUMS requires an accurate GPS system. The accuracy required compared to what is commercially available needs further study.

5. A HUMS provides the opportunity to determine the true flight spectrum of each aircraft. This leads to a number of benefits for the helicopter operator and the flying public. As evidenced from the GCM, ASHM, and UMMC programs, the benefits include the extension of component retirement times, which should result in reduced operating cost

and increased availability, and the ability to better understand how the helicopter is actually used in service, which leads to enhanced safety.

6. Based on the damage tolerance investigation, safe-life components can generally be theoretically redesigned to meet damage tolerance requirements. The component weight increases ranged from 4.6% to 22% for the theoretical redesign of three of the primary structural elements. It is anticipated that for an optimized new design, this weight increase would probably be less. For components with high steady loading and significant oscillatory loading, such as the main rotor yoke, composite materials can potentially be used to meet damage tolerance requirements. It should be noted that, as with safe-life analysis, by better understanding the flight spectrum of each aircraft, the damage tolerance analysis can also be enhanced with the use of a HUMS.

9. REFERENCES.

1. Cronkhite, J., Dickson, B., Martin, W., and Collingwood, G., "Operational Evaluation of a Health and Usage Monitoring System (HUMS)," Report NASA/CR-1998-207409, DOT/FAA/AR-97/64, Army ARL-CR-420, April 1998.

2. Dickson, B., Cronkhite, J., Bielefeld, S., Killian, L., and Hayden, R., "Feasibility Study of a Rotorcraft Health and Usage Monitoring System (HUMS): Usage and Structural Life Monitoring Evaluation," Report NASA CR 198447, DOT/FAA/AR-95/9, February 1996.

3. FAA Advisory Circular, "Airworthiness Approval of Rotorcraft Health and Usage Monitoring Systems (HUMS)," July 15, 1999 (Advisory Circular material will be incorporated in AC 27-1 and AC 29-2).

4. Chang, J.B., Szamossi, M., and Liu, K-W, "A User's Manual for a Detailed Level Fatigue Crack Growth Analysis Computer Code, Volume 1 – the CRKGRO Program," Report AFWAL-TR-81-3093, Volume 1, November 1981.

5. Michael, J., Dickson, B., and Campbell, T., "An Analysis of Use of Damage Tolerant Criteria for Helicopter Design," presented at the *American Helicopter Society Annual Forum*, Montreal, Canada, May 1999.

6. Taylor, D., "A Compendium of Fatigue Thresholds and Growth Rates, Engineering Materials," Advisory Services LTD., Watley, West Midlands, UK, 1985.

7. BHTI Report 412-910-001, "Flight Loads Survey for the Model 412 Helicopter," December 15, 1980.

8. BHTI Report 412-910-006, "Fatigue Life Substantiation for the Dynamic Components of the Model 412 Helicopter," Revision AB, December 21, 1998.